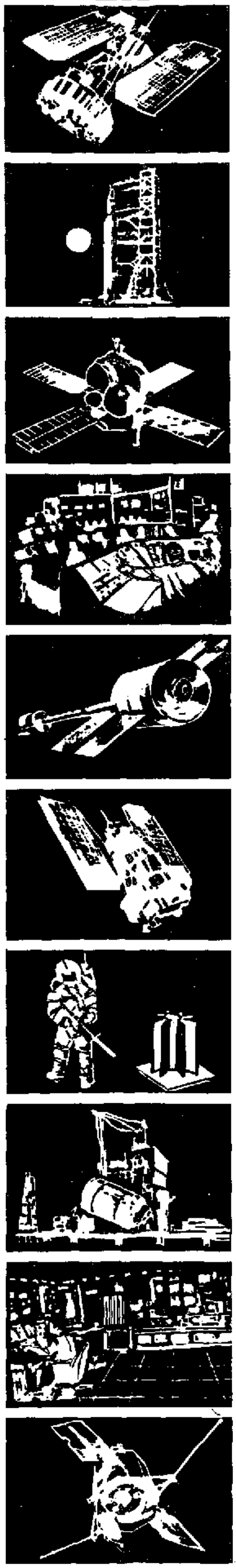
Document No. 74SD4249
16 September 1974

SPACE DIVISION



EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

Report No. 5
SYSTEM DESIGN AND SPECIFICATIONS

Volume 4
MISSION PECULIAR SPACECRAFT SEGMENT AND MODULE SPECIFICATIONS

EARTH OBSERVATORY SATELLITE

Prepared for:
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Under
Contract No. NAS 5-20518

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GENERAL ELECTRIC

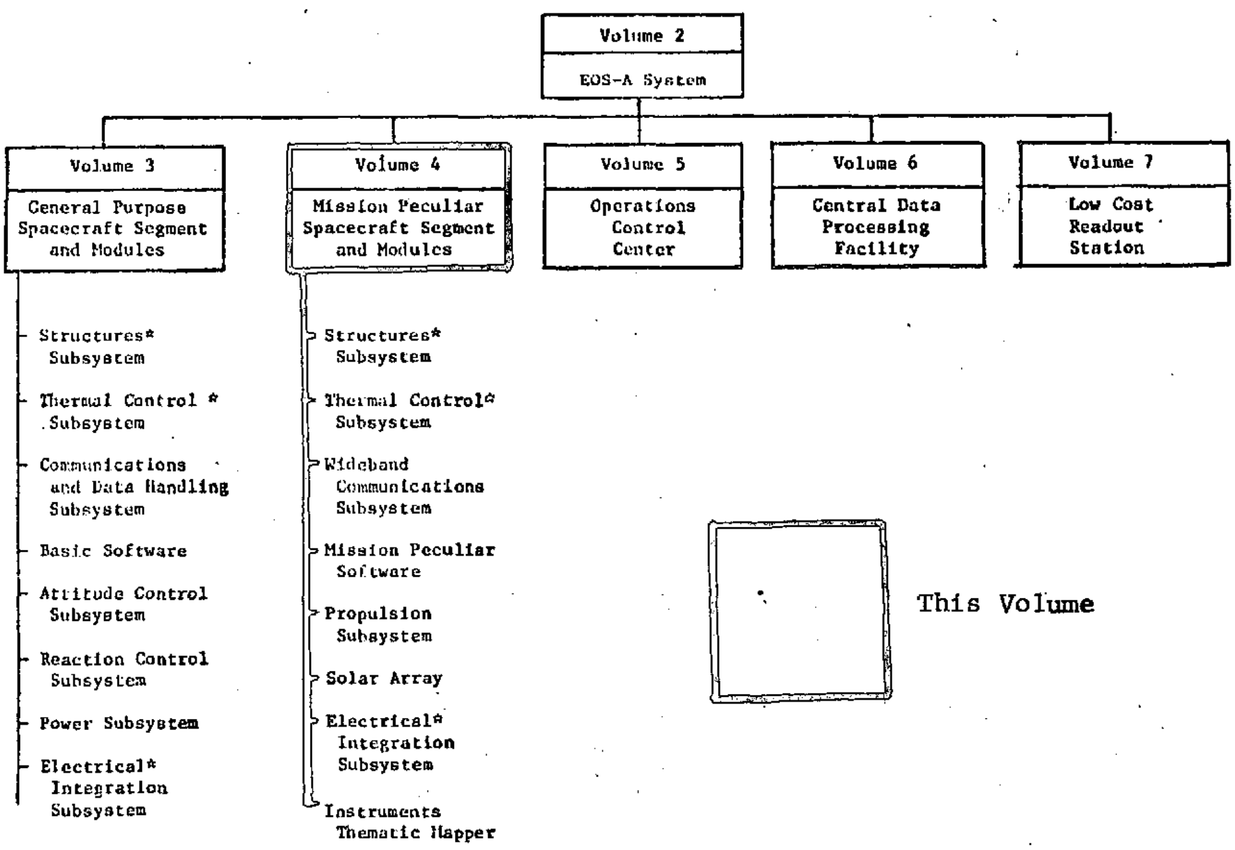
PREFACE

This report, "Baseline System Design & Specifications", has been prepared for NASA/GSFC under contract NAS 5-20518 EOS System Definition Study. It describes the system design that has evolved through a series of design/cost tradeoffs to satisfy a spectrum of mission/system requirements. The basic spacecraft design is compatible with many missions. The EOS-A mission, the potential first mission, is used to define the mission peculiar elements of the system.

For convenience this report is bound in separate volumes as follows:

Volume 1	Baseline System Description
Volume 2	EOS-A System Specification
Volume 3	General Purpose Spacecraft Segment and Module Specifications
Volume 4	Mission Peculiar Spacecraft Segment Specification
Volume 5	Operations Control Center Specification
Volume 6	Central Data Processing Facility Specification
Volume 7	Low Cost Ground Station Specification

Volume 1 "Baseline System Description" presents the overall EOS-A system design, a description of each subsystem for the spacecraft, and the major ground system elements. Volumes 2 through 7 present the specifications for the various elements of the EOS system and are organized according to the specification tree as follows:



* These specifications are written as integral specifications for the GPSS and MPSS and appear in Volume 3 only.

REPORT NO. 5 BASELINE SYSTEM DESIGN & SPECIFICATIONS

VOLUME 4 MISSION PECULIAR SPACECRAFT SEGMENT

SECTION 1
INTRODUCTION

This volume presents the Specifications for the EOS Mission Peculiar Spacecraft Segment and its associated subsystems and modules.

The subsystem and module specifications are presented as "stand-alone" items in order to facilitate their use in further NASA considerations of hardware implementation phases of the EOS program.

It should be noted specifically that reference is made in the MPSS specification to the Structure, Thermal Control and the Electrical Integration Subsystem Specifications. However, in the interest of not being repetitious, these specifications are included only in Volume 3, "Specification for the EOS General Purpose Spacecraft Segment and Modules" of Report No. 5".

The specifications contained in this volume and the sections in which they appear are:

SECTION	TITLE
2.0	SPECIFICATION FOR THE EOS MISSION PECULIAR SPACECRAFT SEGMENT AND MODULES
3.0	SPECIFICATION FOR THE EOS STRUCTURES SUBSYSTEM (VOL 3)
4.0	SPECIFICATION FOR THE EOS THERMAL CONTROL SUBSYSTEM (VOL 3)
5.0	SPECIFICATION FOR THE EOS WIDEBAND COMMUNICATIONS SUBSYSTEM MODULE
6.0	SPECIFICATION FOR THE EOS MISSION PECULIAR SOFTWARE
7.0	SPECIFICATION FOR THE EOS HYDRAZINE PROPULSION SUBSYSTEM MODULE
8.0	SPECIFICATION FOR THE BASIC EOS SOLAR ARRAY ASSEMBLY
9.0	SPECIFICATION FOR THE EOS ELECTRICAL INTEGRATION SUBSYSTEM (VOL 3)
10.0	SPECIFICATION FOR THE SCANNING SPECTRAL RADIOMETER (THEMATIC MAPPER)

Specification No. SVS-XXXX
16 September 1974

SECTION 2.0

SPECIFICATION

FOR THE

EARTH OBSERVATORY SATELLITE (EOS-A)

MISSION PECULIAR SPACECRAFT SEGMENT

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1.0 SCOPE

This specification establishes the requirements for performance and design, qualification and acceptance testing of the Earth Observatory Satellite Mission Peculiar Spacecraft Segment. The Mission Peculiar Spacecraft Segment in conjunction with a General Purpose Spacecraft Segment forms a complete specification for EOS-A. This spacecraft is used to support emerging operational Earth Resource Satellites. It also is used in the development and evaluation of the systems, methods and procedures for collecting, processing and disseminating data from earth orbiting satellites in the broad areas of agriculture and forestry; geology and mineral resources; hydrology and water resources; geography, cartography and cultural resources; and oceanography and marine resources.

The Mission Peculiar Spacecraft Segment consists of those elements of a spacecraft system which are unique to the mission to be performed. The elements generally will be instruments, sensors, communications subsystems for the payload, antennas, structure, thermal control and power elements required for the particular mission.

2.0 APPLICABLE DOCUMENTS

The following documents of the exact issue shown, form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and the detail requirements in the following sections, the detail requirements of this specification shall supersede. In the event of conflict between documents referenced here and lower tier references to documents referenced here, the former supersede.

2.1 APPLICABLE DOCUMENT

SPECIFICATIONS

National Aeronautics and Space Administration

EOS-410-02	Specifications for EOS System Definition Studies, 13 September 1974
S-311-P-11	Quality Monitoring of Integrated Circuits, 1 June 1970
S-323-P-10	Connectors, Subminature Electrical and Coaxial Contacts for Space Flight Use, Revised December 1969

MILITARY

MIL-C-38999	Connectors, Electrical, Miniature, Quick Disconnect, Est. Reliability
MIL-C-29012A	Connectors, Coaxial, RF, General Specification for
MIL-C-26482	Connectors, Electric, Circular, Miniature, Quick Disconnect
MIL-C-17	Cables, RF, Coaxial, Dual Coaxial, Twin Conductors, Twin Head
MIL-W-18044	Wire, Electric Cross-linked, Polyalkene, Insulated, Copper
MIL-E-5400K	Electronic Equipment, Airborne, General Specification for

General Electric

SVS XXXX	Specification for EOS Wideband Communications Subsystem Module
SVS XXXX	Specification for EOS Power Subsystem Module and solar Array
SVS XXXX	Specification for EOS Attitude Control Subsystem Module
SVS XXXX	Specification for EOS Propulsion Subsystem Module
SVS XXXX	Specification for EOS Structural Subsystem
SVS XXXX	Specification for EOS Thermal Subsystem
SVS XXXX	Specification for EOS Mission Peculiar Software
SVS XXXX	Specification for EOS Electrical Integration Subsystem

STANDARDS

National Aeronautics and Space Administration

Aerospace Data System Standards, Part III, Associated Standards, Section I "Radio Frequency and Modulation Standard for Space-to-ground Telemetry", prepared by GSFC Data Systems Requirements Committee, November 1965.

Aerospace Data Systems Standard, Part I, Telemetry Standards, Section I, Pulse Code Modulation Telemetry Standard, prepared by GSFC Data Systems Requirements Committee, January 27, 1966.

Part III, Associated Systems Standards, Section 3 "Spacecraft Minitrack Signal Source Standard", prepared by GSFC Data Systems Requirements Committee, October 1963.

MILITARY

MS33540C	Safety Wiring, General Practices for
MIL-STD-454B	Standard General Requirements for Electronic Equipment
MIL-STD-143A Change 1	Specification and Standards, Order of Precedence for selection of
MS-33586A	Metal, Definition of Dissimilar
MIL-STD-130C	Identification Marking of US Military Property
MIL-STD-1247A	Identification of Pipe, Hose, and Tube Lines for Aircraft, Missile and Space Systems

OTHER PUBLICATIONS

National Aeronautics and Space Administration

NHB 5300.4 (3A) May 1968	Requirements for Soldered Electrical Connections
PPL-12 Latest Issue	GSFC Preferred Parts List
NHB 5300.4 (1A)	Reliability Program Provisions for Space Systems Contractors
NHB 5300.4 (1B)	Quality Assurance Program Provisions for Space Systems Contractors

Air Force Manuals

AFM 71-4	Air Force Regulations for Transportation of Explosive and Other Dangerous Material
AFWTRM127-1	Air Force Western Test Range Safety Manual

MILITARY HANDBOOKS

MIL-HDBK-5A	Metallic Materials and Elements for Aerospace Vehicle\Structure
MIL-HDBK-17	Plastics for Flight Vehicles

INTERSTATE COMMERCE COMMISSION

T.C. George's Tariff No. 6C	Transportation of Explosives and Other Dangerous Articles by Commercial Aircraft
T.C. George's Tariff No. 19	Transportation of Explosives and Other Dangerous Articles by Land, Water, in Rail Freight Service and by Motor Vehicle (Highway) and Water

GENERAL ELECTRIC COMPANY

XXXXX	EOS Mission Peculiar Spacecraft Segment Quality Program Plan
XXXXX	Configuration Management Plan for EOS Mission Peculiar Spacecraft Segment
XXXXX	Reliability Program Plan, EOS Mission Peculiar Spacecraft Segment

3.0 REQUIREMENTS

3.1 General

3.1.1 Mission Objectives

The EOS Mission Peculiar Spacecraft Segment is designed to provide elements of a spacecraft system consisting of the payload and all supporting functions for the payload, including structure, thermal control, electrical distribution, solar array requirements and payload unique communications. The Mission Peculiar Spacecraft Segment combined with an EOS General Purpose Spacecraft provides a total spacecraft system to support a particular mission objective.

3.1.2 Basic Requirements

The EOS Mission Peculiar Spacecraft Segment is designed to meet the following basic requirements:

a) Provide a basic Structural/Mechanical subsystem to support the Mission Peculiar modules including instruments throughout the launch and orbit environment.

b) Provide a Thermal Control System which is self efficient and will not require or cause interaction with the General Purpose Spacecraft segment of the satellite.

c) Provide a Solar Array which is capable of supplying all power required for the operation of the basic General Purpose Spacecraft Segment as well as power to the mission peculiar payload.

d) Provide a Wideband Communications Subsystem Module which is capable of retrieval of payload data via STDN, TDRS and low cost user data links.

e) Provide a Propulsion Subsystem Module which is capable of performing the spacecraft functions of orbit adjust and orbit transfer.

3.1.3 Orbit Design Restraints

The EOS Mission Peculiar Spacecraft Segment must be capable of performing in earth orbits ranging from a minimum of 250 n.m. to 500 n.m. Nominal orbit will be 418 n.m. circular sun-synchronous, 1100-1130 descending node.

3.1.4 System Relationship

The Mission Peculiar Spacecraft Segment is part of a total earth orbiting satellite system in conjunction with a General Purpose Spacecraft and is the segment of the satellite that houses the payload and payload supporting functions. Along with an Operations Control Center, a Data Processing Facility and attendant Launch Vehicle Operations, the Mission Peculiar Spacecraft Segment makes up a total program system which is capable of being varied considerably from mission to mission. The modular/flexible design of the Mission Peculiar Spacecraft Segment allows it to perform this function in a large variety of cases.

3.2 SPACECRAFT SYSTEM PERFORMANCE

3.2.1 Spacecraft Service Subsystems

3.2.1.1 Structural Subsystem

The EOS Mission Peculiar Spacecraft Segment modular design is the "payload" section of a total spacecraft assembly. The Mission Peculiar segment consists of a core structure supporting the Wideband Communications Module, the Instrument Modules (Thematic Mapper and Multispectral Scanner) and the TDRSS antenna. Also considered to be Mission Peculiar are the Solar Array, Solar Array Drive, the orbit adjust and orbit transfer functions of the Propulsion Module and the Launch Vehicle Adapter. The Instrument Section structure provides the capability to support a maximum of 500 pounds of payload equipment.

3.2.1.2 Thermal Subsystem

The thermal environment is passively controlled to provide a temperature range of 70 $\pm 5^{\circ}$F for the Instrument, Wideband Communications Subsystem, and Propulsion Modules. Each module is thermally independent and is designed not to interact with the thermal response of the other modules or subsystem structure.

3.2.1.3 Communications and Data Handling Subsystem Module

This subsystem is specified in the General Purpose Spacecraft Segment Specification. No mission peculiar modifications are required for EOS-A.

3.2.1.4 Wideband Communications Subsystem Module

The EOS-A Wideband Communication Subsystem accepts, processes, and transmits data in real time from a Thematic Mapper (TM) and a Multispectral Scanner (MSS) sensor. Four independent spacecraft-to-ground R.F. links are provided; a link via TDRS which gives extra-continental coverage for TM & MSS data with a Thematic Mapper Compacted (TMC) data back-up, two identical STDN links for TM & MSS with a TMC back-up, and a Low Cost User (LCU) link for either MSS or TMC data. The subsystem will be compatible with follow-on mission sensors, provide for shuttle retrieval but not service and enable back-up operational modes.

Table 3.2.1.4-1 lists frequencies, bandwidths and modulation for the assigned data links.

Table 3.2.1.4-1. Wideband Module Frequency Applications

Data Link	Use	Nominal Frequency GHz	Allocated Bandwidth MHz	Modulation
EOS to TDRSS	TM & MSS Data	15.004	225	QPSK/FM
EOS to STDN	TM & MSS Data	8.150	310	QPSK/FM
EOS to LCRS	MSS Data Compacted TM Data	8.340	30	FM

3.2.1.5 Attitude Control Subsystem Module

This subsystem is specified in the General Purpose Spacecraft Segment Specification. No mission peculiar modifications are required for EOS-A.

3.2.1.6 Power Subsystem Module and Solar Array

The Power Subsystem Module is part of the General Purpose Spacecraft Segment Specification (SVS XXXX). The Solar Array is considered to be Mission Peculiar and is therefore part of this specification. The Power Subsystem Module and the Solar Array shall be designed to permit normal operation of the spacecraft subsystems as well as payload operations. The power subsystem capability shall not restrict payload operation, either real time or store and playback, under the following constraints:

a. Data Acquisition Stations specified in paragraph TBD with 5^{o} elevation or local terrain masking, whichever is greater.

b. Payload power requirements of 200 watts (orbit average)

c. Nominal earth orbit of 418 n.m. (775 Km)

d. All payload and service subsystems operating normally at the end of one year.

e. Average payload operating time of 15 minutes per orbit during sunlit part of orbit (i.e. globe coverage).

The spacecraft solar arrays shall be controlled to a tracking error of ±10 degrees during the day portion of each orbit, and shall settle to this accuracy within 1/2 of one orbit from paddle deployment or initiation of tracking.

The tracking angle is defined as that angle between the plane containing the paddle axis and the sunline and the plane containing the paddle axis and paddle perpendicular. During the dark portion of the orbit, the paddles shall continue to rotate at orbital rate ±.3 degree per minute.

3.2.1.7 Propulsion Module

The Propulsion Module has the combined capability of performing the spacecraft functions of reaction control, orbit adjust, and orbit transfer. This specification addresses only the orbit adjust and orbit transfer sections of the Propulsion Module. Reaction control is specified in the General Purpose Spacecraft Segment Specification.

The Propulsion Module must provide the following mission peculiar functions:

- Orbit Adjust Functions

Inject. Error Removal - In Plane	42 FPS
- Cross Track	16.5 FPS
Orbit Maintenance	1.4 FPS/Yr.

- Orbit Transfer Functions

Mission Orbit Establishment	Not Required
Retrieval - 300 Nm Circ.	190.7/192.2 FPS
- 250 Nm Circ. (Alternate)	273.8/276.9 FPS
S/C Control	100% Duty Cycle for one engine
Velocity Trim	Not Required

3.2.2 Payload Subsytems

3.2.2.1 Thematic Mapper Subsystem

The Thematic Mapper requirements are specified in GSFC Specification (73-15012A) for an Earth Observation Scanning Spectro-Radiometer Experiment.

Significant Parameters of the EOS Thematic Mapper are tabulated below:

Altitude		775 kilometers
Number of Spectral Bands		6
IFOV	Band 1 to 5	35 μ radians
	Band 6	140 μ radians
Detectors/strip	Band 1 to 6	16
	Band 7	4
Size (envelope)		48" x 40" x 40"
Thematic Mapper		330 #

3.2.2.2 Multispectral Scanner (MSS)

The MSS interface is specified in Design Study Specification for the Earth Resources Technology Satellite, ERTS A and B, S-701-P-3, Attachment III, Interface Characteristics Four Band Multi-Spectral Scanner, and GSFC Specification for the Multispectral Scanner System, S-731-P-100A, January 1970. (To be updated for Five Band System.)

3.2.2.3 Data Collection System (DCS)

The Spacecraft DCS equipment receives transmissions from the Data Collection Platforms and transfers it to the C&DH Narrow Band Telemetry Subsystem for retransmission, via the USB Transmitter. The C&DH Module shall provide for the DCS receivers and antenna whose characteristics are as follows:

Receiver Size:	TBD
Receiver Weight:	TBD
Power Required:	TBD
Telemetry Requirements	TBD
Command Requirements	TBD
Antenna Size	TBD
Antenna Field of View	TBD
Receiver Electrical:	TBD

Output Frequency
Bandwidth
Output Signal
Equiv. RMS Test Tone
RMS Noise
Input Frequency
Bandwidth:
Dynamic Range:

TBD

The spacecraft equipment shall be compatible with the Data Collection Platforms interface requirements specified below:

The Data Collection Platforms (DCP's) shall have the following characteristics:

Transmission Frequency:

Bandwidth:

Modulation/Deviation:

Minimum ERP:

Signal Level at Receiver:

TBD

3.2.3 AGE Subsystems

3.2.3.1 Transportation

The Spacecraft Transportation Equipment provides for protection of the Mission Peculi Spacecraft Segment during transportation over the road or by air. Within its container, the spacecraft will not be subjected to environments in excess of those specified in Paragraph 3.4.4.

3.2.3.2 Fuel Servicing

The fuel service equipment will provide for loading 225 lbs. of Hydrazine and 0.6 lbs. of GN_2 at 375 psi for orbit adjust, maintenance, and transfer systems.

3.2.3.3 Mass Property

Mass Property measurement equipment will provide capability for (a) locating the c.g. within $\pm$ 0.020 inches (x and y axes) and $\pm$ 0.050 inches z (axis), (b) determining the weight to within $\pm$ 0.1% and (c) determining the products of inertia to within the following:

I_{xz}	(roll-yaw)	0.85 slug-ft.^{2}
I_{yz}	(pitch-yaw)	3.0 slug-ft.^{2}
I_{xy}	(roll-pitch)	No limit

3.2.3.4 Alignment

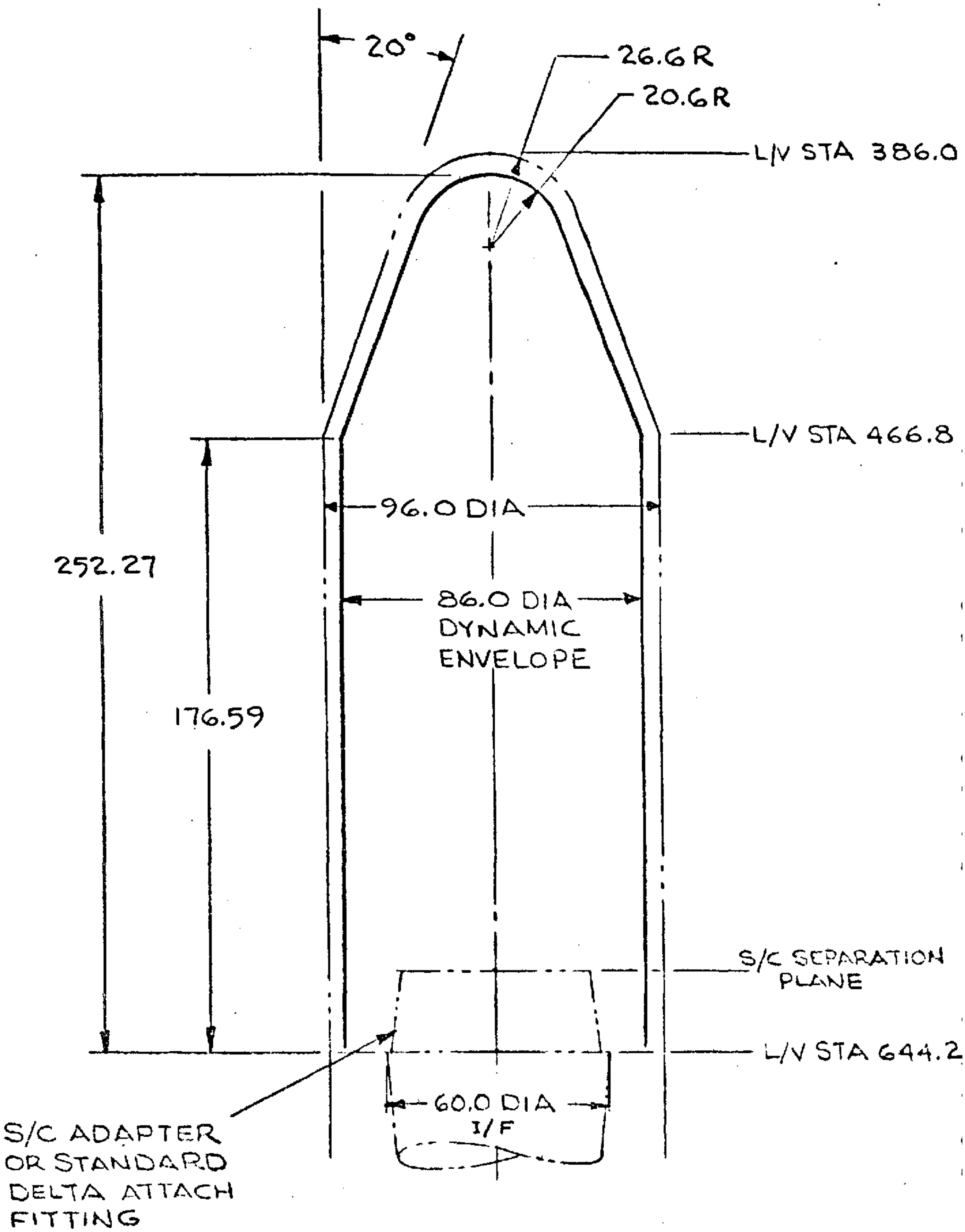
Alignment equipment will provide capability to permit alignment of mission peculiar spacecraft mounted equipment as specified in paragraph 3.5.1.12.

3.2.3.5 Electrical Test

Electrical test equipment will provde capability to control and monitor spacecraft system and subsystem operation during system and subsystem levels of testing described as specified in paragraphs TBD. The equipment shall be designed to protect the spacecraft from damage in the event of a malfunction in the AGE. Control of instrument targets shall be provided.

3.2.3.6 Ground Station

A test ground station will provde capability through both STDN and TDRSS links, to receive, process, record and display wideband communications data.



DELTA LAUNCH VEHICLE
FAIRING ENVELOPE

FIGURE 3.3.1.2-1

3.2.3.7 Environmental Support

Equipment necessary to supply power and control heater arrays and temperature monitoring equipment will be provided for vacuum thermal testing.

3.3 SYSTEM INTERFACE REQUIREMENTS

3.3.1 Launch Vehicle

The Mission Peculiar Spacecraft Segment plus a General Purpose Spacecraft Segment will be launched into orbit by the Delta 2910 Launch Vehicle and shall be compatible with the launch vehicle interfaces indicated in the following paragraphs. The overall spacecraft will also be capable of being retrieved by the Shuttle.

3.3.1.1 Flight Loads

The Mission Peculiar Spacecraft Segment shall withstand the launch vehicle induced flight loads as given in Section 3.4.4.

3.3.1.2 Mechanical Interface

The Mission Peculiar Spacecraft Segment plus a General Purpose Spacecraft shall fit within the shroud/spacecraft envelope shown in Figure 3.3.1.2-1. These clearances shall be maintained during static conditions and while the spacecraft is subjected to the flight loads defined in Section 3.4.4.

3.3.1.3 Access

The launch vehicle shroud shall provide access doors for several (TBD) spacecraft ground functions as required.

3.3.1.4 Electrical Interface

3.3.1.4.1 Ground Monitoring (Overall Spacecraft)

The launch vehicle/spacecraft interface shall provide wiring and connectors for up to ___ pins for hardwire ground monitoring of spacecraft functions from installation of the spacecraft until liftoff. This wiring shall be routed from the block house to the launch vehicle ground umbilical and up through the spacecraft adapter-launch vehicle transision ring.

3.3.1.4.2 Interface Pin Assignments

The launch vehicle/spacecraft interface shall provide for ground monitoring and launch phase telemetry, electrical power and event functions. Wiring and connectors for these functions shall be fed through the spacecraft adapter-launch vehicle transition ring.

3.3.1.4.3 Electroexplosive Device (EED) Requirements

Launch vehicle shall provide power to fire four electroexplosive bolt cutters to effect spacecraft separation. 150% all-fire current shall be provided. The EED's have the following characteristics:

a. Recommended Fire Current: 5.0 amperes minimum for 20 milliseconds per squib*

b. No-fire Currents: 1.0 amperes maximum for 5 minutes per squib*

c. Bridgewire Resistance: 0.25 + 0.1 ohms per squib*

d. All-fire Current: 2.58 amperes minimum per squib*

* Per squib means per bridgewire.

3.3.1.4.4 RF Transmissibility

Should RF Mission Peculiar communications links (as shown in Table 3.3.1.4-4) between the spacecraft and the NASA telemetry area at the launch site be necessary during the time the spacecraft is installed on the launch vehicle at the launch pad, re-radiator antenna will be provided and hardwired to the telemetry area or the shroud will be modified with windows. If the windows are used, the losses through the shroud should be less than 3 for frequencies between 30 and 2300 MHz.

Table 3.3.1.4-4 S/C RF Signature

Function	Nominal Frequency GHz
TDRSS & MSS Data	15.005
STDN - TM & MSS Data	8.150
LCRS - Compacted TM, MSS Data	8.340

3.3.1.5 Environmental Control

3.3.1.5.1 Thermal Control

The launch vehicle shall be sealed at the spacecraft interface to prevent air flow from the launch vehicle to the spacecraft. Cooling air at an inlet temperature of 50 to 66°F shall be provided within the shroud to control the spacecraft environment on the launch pad to lift-off. Cooling air flow rate shall be at least 50 pounds per minute. The relative humidity of the cooling air shall not exceed 50%. The spacecraft shall be provided an environmental equal to or better than class 100,000 at all times.

3.4 SYSTEM DESIGN REQUIREMENTS

3.4.1 Reliability

The Mission Peculiar Spacecraft Segment is designed to optimize system reliability. Functional and block redundancy are utilized to enhance system reliability.

3.4.2 Maintainability

The Mission Peculiar Spacecraft Segment shall be designed for ease of maintainability to minimize downtime during assembly, test, and check-out.

3.4.2.1 Maintenance Requirements

a. The Mission Peculiar Spacecraft Segment shall be designed for removal and replacement down to the module level.

b. All electrical parameter trimming, mechanical adjustment or alignment performed at the subsystem or lower level shall be performed prior to subsystems Flight Acceptance testing and shall not be subsequently changed without recertifying the subsystem.

c. Spacecraft system level electrical trimming, mechanical adjustment or alignment shall be such that these parameters can be re-established in the event of subsequent disassembly and assembly of the spacecraft.

3.4.2.2 Maintenance and Repair Cycles

The Mission Peculiar Spacecraft Segment shall be designed such that no scheduled maintenance will be required after the hardware has been shipped to the launch site with the exception of the normal servicing functions associated with propellant loadings, and Electroexplosive Device (EED) installations. Repairs to be effected at the launch site shall be limited to those failures or malfunctions which are discovered at the launch site and shall only be accomplished by replacement of equipment at the module level.

a. Access shall be made to all module test plugs, harness break-in points, and pressurant and propellant fill and drain valves through openings, access ports or doors without disassembly of the spacecraft.

b. Optical references located on critically aligned components shall be externally visible through inspection ports or access doors.

c. Access shall be provided for normal servicing before installation of thermal blankets for such items as EED removal and installation and the capping of critical optical or pneumatic components.

d. Access shall be provided for installation and subsequent removal of all non-flight hardware before launch.

3.4.3 Useful Life

The useful life of the spacecraft shall be a minimum of 1 year starting with the acceptance of the spacecraft by the procuring agency. The useful life of individual subsystems and modules within the spacecraft shall include sufficient additional time to allow for the elapsed time during transportation, handling, storage and testing phases prior to acceptance by the procuring agency. The orbital operational life shall be a minimum of two years.

3.4.4 Environmental

The spacecraft shall suffer no performance degradation beyond the limits specified elsewhere in this document while exposed or after exposure to the environmental conditions specified in Table 3.4.4-1. This table lists the environmental requirements for each phase of spacecraft life from transportation through orbital operation plus environmental conditions to which the equipment shall be exposed during Qualification and Acceptance Tests. Environmental criteria supplementary to Table 3.4.4-1 is contained in Sections 3.4.4.1 through 3.4.4.5.

The Mission Peculiar Spacecraft Segment will operate during portions of Handling and Assembly, Test and Checkout, Pre-launch, Launch, and Orbital phases. The spacecraft will not operate during the Transportation phase.

3.4.4.1 Shipping, Handling, and Transportation

3.4.4.1.1 Acceleration

During the shipping and handling, equipment shall be capable of experiencing limit loads of up to ±3 g in any direction.

3.4.4.1.2 Vibration

Equipment in its shipping containers shall be capable of withstanding the following sinusoidal vibration environments:

Frequency (Hz)	Acceleration (g; 0-Peak)	Displacement (Inches, D.A.)
2-5	±0.375	
5-1000	±1.3	0.30

3.4.4.1.3 Shock

Equipment in its shipping containers shall be capable of withstanding a 3 G maximum acceleration in any axis.

3.4.4.2 Structure Performance

PHASE ENVIRONMENT	X-PORT	HANDLING & ASSEMBLY/TEST & CHECKOUT	PRE-LAUNCH	LAUNCH	ORBITAL	ENVIRONMENTAL TESTING QUAL.	ACCEPT.
THERMAL (°F)	+35 to +100	72 ±3 (Controlled) 65 ±15 (Uncontrolled)	50 to 66	(TBD) INSIDE SHROUD SURFACE 200	MSF:440±13 Albedo 167 Earth Emitt.67 (BTU/HR/FT^2)	N/A	See Figure 3.4.4-3
PRESSURE (mm Hg)	300 to 790	760 ± 30	760 ±30	780 to 1X10^{-18}	1X10^{-18}	790 to 1X10^{-5}	790 to 1X10^{-5}
HUMIDITY %RH	55	45 to 55	35 to 55	NA	NA	Not Req'd	Not Req'd
VIBRATION	See Para. 3.4.4.1.2		NA	See Para. 3.4.4.2.2	NA	See Para 3.4.4.2.3	See Para 3.4.4.2.3
SHOCK	See Para. 3.4.4.1.3		NA	See Para. 3.4.4.5	NA	See Para 3.4.4.4	See Para 3.4.4.4
ACCELERATION	See Para. 3.4.4.1.1	See Para. 3.4.4.1.1	NA to Z Axis	See Para. 3.4.4.2	TBD	Not Req'd	Not Req'd
ACOUSTIC NOISE	NA	NA	NA	See Para. 3.4.4.4		See Para 3.4.4.3	See Para 3.4.4.3
PARTICLE BOMBARD	NA	NA	NA	Table 3.4.4-2	Table 3.4.4.2	Not Req'd	Not Req'd
METEROID	NA	NA	NA	NA	Table 3.4.4-3	Not Req'd	Not Req'd

Table 3.4.4-1 Environmental Criteria

3.4.4.2.1 Support

The Mission Peculiar Spacecraft Segment shall be capable of supporting a maximum gross weight of $\approx$ 1280 pounds with a weight breakdown as shown in Table 3.4.4.2.1-1.

Table 3.4.4.2.1-1

Mission Peculiar Spacecraft Segment Weight Breakdown

	S/S	TOTALS
MISSION PECULIAR S/C SEGMT.		762
ORBIT ADJ. & O.T.	190	
TDRSS	75	
WIDE BAND COM.	134	
MISS. PECULIAR & MECH.	185	
MISS. PEC. HARNESS/T.C.	64	
SOLAR ARRAY & DRIVE	114	517
PAYLOAD INSTRUM.		
TM	350	
MSS	155	
DCS	12	

3.4.4.2.2 Stiffness

For powered flight, the primary and secondary structures shall provide adequate stiffness to satisfy the minimum resonant frequency requirements in Table 3.4.4.2.2-1. For this evaluation, the spacecraft and modules shall be analyzed in the launch configuration cantilevered from their attachment points.

In the orbital configuration, a TBD Hertz minimum resonant frequency for appendages shall be adequate to preclude dynamic interaction between the structure and the Attitude Control System. Analytical evaluations of the deployed solar arrays, including in detail the effects of hinge flexibility and solar drive flexibility, shall be used to demonstrate compliance with this requirement.

3.4.4.2.3 Strength

A. For preliminary sizing, the primary structure shall be designed to the qualification level steady-state accelerations of Table 3.4.4.2.3-1. Off-loading conditions for the Delta configuration shall be considered in the analysis. Subsequent dynamic analyses shall determine responses to the qualification vibration test levels of Tables 3.4.4.2.3-2, and 3.4.4.2.3-3, including estimated "notching" levels to prevent excessive dynamic test loads. In these response analyses, a modal damping ratio of $C/C_c = 0.05$ will be used for the primary structure modes.

Table 3.4.4.2.1-1

Spacecraft Minimum Resonant Frequencies

During Powered Flight

A. Primary Structure

Launch Vehicle / Structure	THOR/DELTA Lat. (Hz)	THOR/DELTA Long. (Hz)	TITAN III B Lat. (Hz)	TITAN III B Long. (Hz)	SHUTTLE Lat. (Hz)	SHUTTLE Long. (Hz)
Spacecraft *	10	30	10	30	TBD	
Subsystem Module **	60	60	60	60	60	60
Experiment ***	TBD		TBD		TBD	

* Cantilevered from the launch vehicle/adapter interface.

** Mounted on ball joints at the four attachment points.

*** Cantilevered from the transition ring attachment points.

B. Secondary Structure

Item	Longitudinal (Hz)	Lateral (Hz)
Instrument Section Structure	70	70
Antenna Mounting	70	70
Stowed Solar Array Module Mounting	25-30	8
Stowed Solar Array Panel	70	15
Subsystem Component Mounting	100	100

Table 3.4.4.2.3-1

Qualification Level Quasi-Steady Accelerations

Launch Vehicle & Condition	Longitudinal (g)	Lateral (g)
Delta		
Max. Lateral (Lift-off)	- 4.4	± 3.0
Max. Compression (MECO/POSO)	-18.0	± 1.0
Max. Tension	1.5	± 3.0
Titan III B/NUS		
Max. Lateral (Lift-off)	- 2.9	± 2.5
Max. Compression (Stage II Shutdown)	-13.5	± 1.3
Max. Tension (Stage I Shutdown)	3.1	± 1.9
Shuttle		
Lift-off	- 3.5	± 1.3
Orbiter End Burn	- 5.0	± 0.6
Entry	0.4	4.5
Landing & Braking	± 2.3	3.8
Crash (Ultimate Applied Separately)	9.0 - 1.5	4.5 - 2.0

Table 3.4.4.2.2-2 Sinusoidal Vibration

Axis \ Launch Vehicle	Delta		Titan III B		Space Shuttle	
	Frequency (Hz)	Acceleration (G, o-pk)	Frequency (Hz)	Acceleration (G, o-pk)	Frequency (Hz)	Acceleration (G, o-pk)
Longitudinal Axes	5-15* 15-21 21-100	2.3 6.0 2.3	5-20* 20-50 50-200	9.0 in/sec 3.0 2.3		TBD
Lateral Axes	5-14* 14-100	2.0 1.5	5-22* 22-200	2.0 1.5		TBD

Sweep Rate: 2 octaves/minute

* Limited with the performance of the exciter. The amplitude in these frequency ranges shall not exceed 0.5 inches D.A.

Table 3.4.4.2.2-3 Random Vibrations*

	Frequency (Hz)	PSD (G^2/Hz)	GRMS	TIME (Seconds Per Axis)
Thor Delta	20-300 300-700 700-2000	+4 db/Oct .16 -3 db/Oct	14.1	20
	20-300 300-700 700-2000	+4 db/Oct .07 -3 db/Oct	9.5	70
Titan III D	20-250 250-2000	+6 db/Oct .16	17.0	240
Shuttle	20-100	+6 db/Oct .65 -6 db/Oct	24.3	90

* Thrust and Lateral Axes

The final design loads shall be determined in the subsequent phase from coupled launch vehicle-spacecraft dynamic response analysis and dynamic analyses of other critical conditions. These analyses results shall be used to determine the notching factor for the sinusoidal vibration tests at the primary spacecraft resonances such that the primary structure stress levels experienced during flight shall be compatible with those experienced during the vibration test.

B. Steady State Accelerations.

a. Hoist and Spacecraft to L/V mating

Nz	Nx	Ny
-3.0	±0.5	±0.5

b. Transportation (Rail, Air, Motor, and Water)

Vertical: 2.0 g up
4.0 g down

Lateral (sideways): ± 2.5 g

Longitudinal: ±3.0 g (due to docking ramp impact)

These loads are maximum expected equivalent static laods due to carrier operation They are to be applied separately. The directional terms are with respect to the carrier motion. These loads are to be reacted by the appropriate transportation support configuration. Special procedures, handling equipment, transportation support configuration and shock and vibration isolation between spacecraft and carrier floor will be utilized in order not to exceed the above loads and that the spacecraft element loadings do not exceed 50 % of the flight qualification loads.

The design load factors of safety shown in Table 3.4.4.2.3-4.applied to qualification loads presented in Table 3.4.4.2.3-1 to obtain the structural design yield and design ultimate loads.

The pressure vessel factors shown in Table 3.4.4.2.3-5 shall be applied to maximum expected operating pressures to obtain design pressures for all hydraulic and pneumatic components.

Table 3.4.4.2.3-4 Design Load Factors of Safety

Load Condition	Design Lead Factors of Safety	
	Yield	Ultimate
Launch (Qualification Level)	1.5	2.0
Orbital (Qualification Level)	1.5	2.0
System Qualification Test	1.5	2.0
Transportation, Handling (Apply Load Factors to Loads of Paragraph 3.2.1.2.3.1	1.5	2.0

Table 3.4.4.2.3-5 Pressure Vessel Factors

Pressure Container	Operating	Proof	Burst
Main Propellant Tanks	1.00	1.50	2.00
Vessels Including Accumulators & Pressurization Bottles	1.00	1.50	2.50
Hydraulic & Pneumatic Lines, Fittings & Hoses	1.00	2.50	4.00
Propellant Supply and Vent Components	1.00	1.50	4.00

The Mission Peculiar Spacecraft Segment when subjected to the environments presented herein shall maintain the minimum design margins of safety presented in Table 3.4.4.2.3-6.

Table 3.4.4.2.3-6 Minimum Margins of Safety

	Minimum Margin of Safety
Fasteners in Shear	+.15
Bolts in Tension	+.50
Fittings	+.15
Lugs	+.25
Welds - Electron Beam	+.15
Welds - Other	+.50 (Dependent on Inspection Procedure)
Bonded Joints	+.50

Margins of safety less than 2.0 shall be indicated numerically. Those greater than 2.0 may be listed as high.

3.4.4.3 Acoustic Levels

The estimated acoustic spectra during flight are presented in Table 3.4.4.4-1.

Table 3.4.4.3-1 Acoustic Levels

Octave Band Center Freq. (Hz)	Sound Pressure Level: db ref. .0002 dynes/cm^2			
	Thor/Delta		Titan III D	Shuttle
31.5	129	124	124	131
63	130	125	130	137
125	134	129	138	141
250	139	134	143	143
500	147	142	142	143
1000	141	136	137	141
2000	138	133	133	138
4000	131	126	130	134
8000	128	123	128	130
Overall	149	144	147	149
Duration (Seconds)	20	70	120	120

3.4.4.4 Shock

T.B.D.

Table 3.4.4-2 Particulate Radiation

Solar High Energy Particle Radiation

Composition: Predominately of protons (H^+) and alpha (He^{++})

Integrated yearly flux:

Energy > 30 Mev	$N \approx 8 \times 10^9$ protons/cm^2 near solar maximum.
	$N \approx 5 \times 10^8$ protons/cm^2 near solar minimum.
Energy > 100 Mev	$N \approx 6 \times 10^8$ protons/cm^2 near solar maximum.
	$N \approx 1 \times 10^8$ protons/cm^2 near solar minimum.

Maximum dosage with shielding of 5 gm/cm^2: $\approx$ 200 rads per week (3 flares)

Table 3.4.4-3 Meteoroid Environment

The encounter frequency (N), in number per square meter per second, of sporadic cometary meteoroids with a mass equal to or greater than m grams on a randomly oriented surface is

$$\log N = -.05426\,(\log m)^2 - 1.614 \log m$$

$$- 14.644 + \log\left\{1 + (0.419/r)\right\}$$

$$+ \log\left\{\left(1 + \sqrt{1 - 1/r^2}\right)/2\right\} + \log F_{seasonal}$$

where r is the distance from earth in units of earth's radius (6.378×10^3 km) and $F_{seasonal}$ is a seasonal factor obtained from the table given below. The seasonal factor is obtained by taking the average of monthly factors listed for the months of the mission duration.

SEASONAL FACTORS

January	.6
February	.4
March	.5
April	.6
May	1.1
June	1.6
July	1.8
August	1.6
September	1.1
October	1.1
November	.9
December	.7

3.4.5 Transportability

The design of the Mission Peculiar Spacecraft Segment and its packaging for shipment shall be such that the spacecraft will meet all performance requirements stated in Section 3 of this specification after the spacecraft in the AGE transportaion equipment is subjected to the transportation environments described in paragraph 3.4.4 of this specification. Transportation shall be by highway and/or air supported in the AGE shipping container on a test and calibration dolly.

Explosive devices shall be shipped separately from the spacecraft for installation at the launch site. These devices shall be packaged and transported in accordance with ICC Tariffs No. 6C (for commercial aircraft) and No. 19 (for other than commercial aircraft.)

3.4.6 Safety

3.4.6.1 Ground Safety

Design consideration shall be given to minimize hazardous interaction of equipment, facilities, and facility equipment during spacecraft manufacture, test, and final installation. Suitable precautions shall be specified in spacecraft handling, assembly, and test instructions. Ground operating procedures shall incorporate warning and cautionary instructions to preclude inadvertent equipment damage or personnel injury resulting from inherent RF radiation, explosive, and pressure vessel hazards. Parts which may work loose in service shall be safety wired in accordance with MS33540, or shall have other approved locking means applied.

3.4.6.2 Personnel Safety

Personnel safety shall be in accordance with MIL-STD-454, Requirement 1. Adequate means for preventing inadvertent deployment of the solar array during ground handling and test operations shall be employed.

3.4.6.3 Explosive and/or Ordnance Safety

Safe-arm plugs shall be used to maintain a short circuit across each ordnance device during assembly and ground handling. Shorting plugs shall be placed in the harness connecting the firing circuit (pyrotechnic controller) to the explosive devices and shall be located in a close proximity to the explosive devices. The shorting plugs shall be replaced by arm plugs during preparation of the spacecraft for launch. The requirements for propellant (hydrazine) and explosive device handling and transfer shall be in accordance with AFWTRM-127-1 and AFM-71-4.

3.5 DESIGN AND CONSTRUCTION

3.5.1 General Design Features

3.5.1.1 Spacecraft Reference Axes

Spacecraft reference axes for determination of mass properties shall be as shown in Figure 3.5.1.1-1.

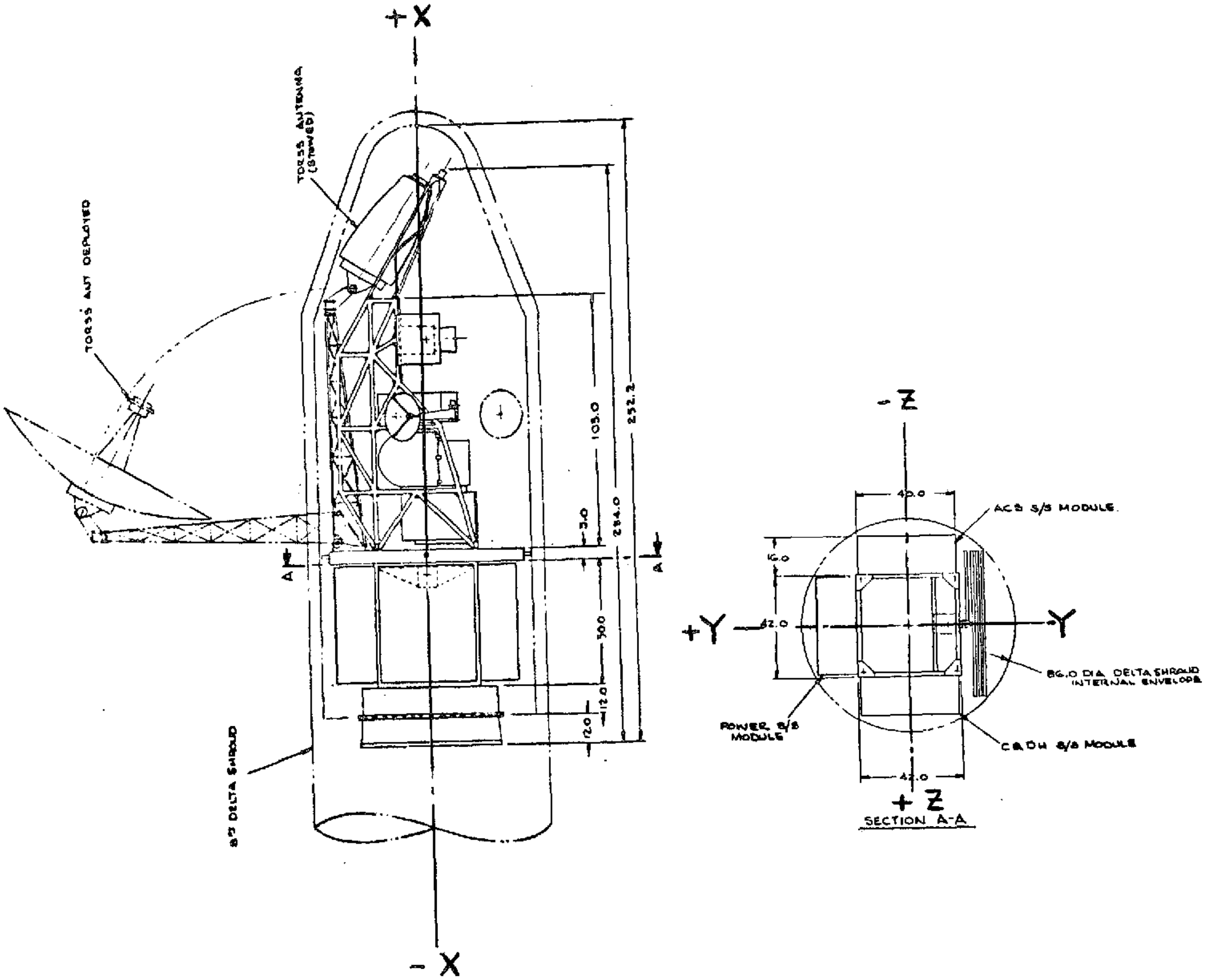


FIGURE 3.5.1-1. EOS SPACECRAFT REFERENCE AXES

3.5.1.2 Mass Property Restraints

3.5.1.2.1 Launch Weight

The Mission Peculiar Spacecraft Segment weight, including the structure, Wideband Module, Solar Array, Instruments, TDRSS antenna, Adapter, harness, thermal control and propulsion shall not exceed 1375 lbs.

3.5.1.2.2 Center of Mass *

The radial offset of the center of mass from the z axis (yaw) shall not exceed 1.0 inches in the separation mode. (Less adapter & array folded).

In the orbit mode (less adapter, array open)the position of the center of mass shall not deviate more than

$x = \pm 0.5$ inches

$z = \pm 0.5$ inches

$y = \pm 0.5$ inches

3.5.1.2.3 Products of Inertia *

In the orbit mode (less adapter, array open) the products of inertia shall not exceed,

Pxz = 0 (spacecraft ballasted)

Pxy = $\pm$ N.A. Sl-Ft.^{2}

Pzy = $\pm$ 3 Sl-Ft.^{2}

3.5.1.2.4 Moments of Inertia *

In the orbit mode (less adpater, array open) the moments of inertia shall not exceed,

Ix = TBD
Iz = TBD
IY = TBD

ORIGINAL PAGE IS
OF POOR QUALITY

* The center of mass, products and moments of inertia requirements exist for a total spacecraft, which includes the General Purpose Spacecraft Segment and a Mission Peculiar Spacecraft Segment.

3.5.1.3 Mission Peculiar Spacecraft Segment Configuration

An exploded view of the Mission Peculiar Spacecraft Segment is shown in Figure 3.5.1.3-1. Each element of Mission Peculiar subystems is illustrated. The General Purpose Spacecraft Segment fits in between the instrument section structure and the Propulsion Module and is illustrated in the figure in phantom.

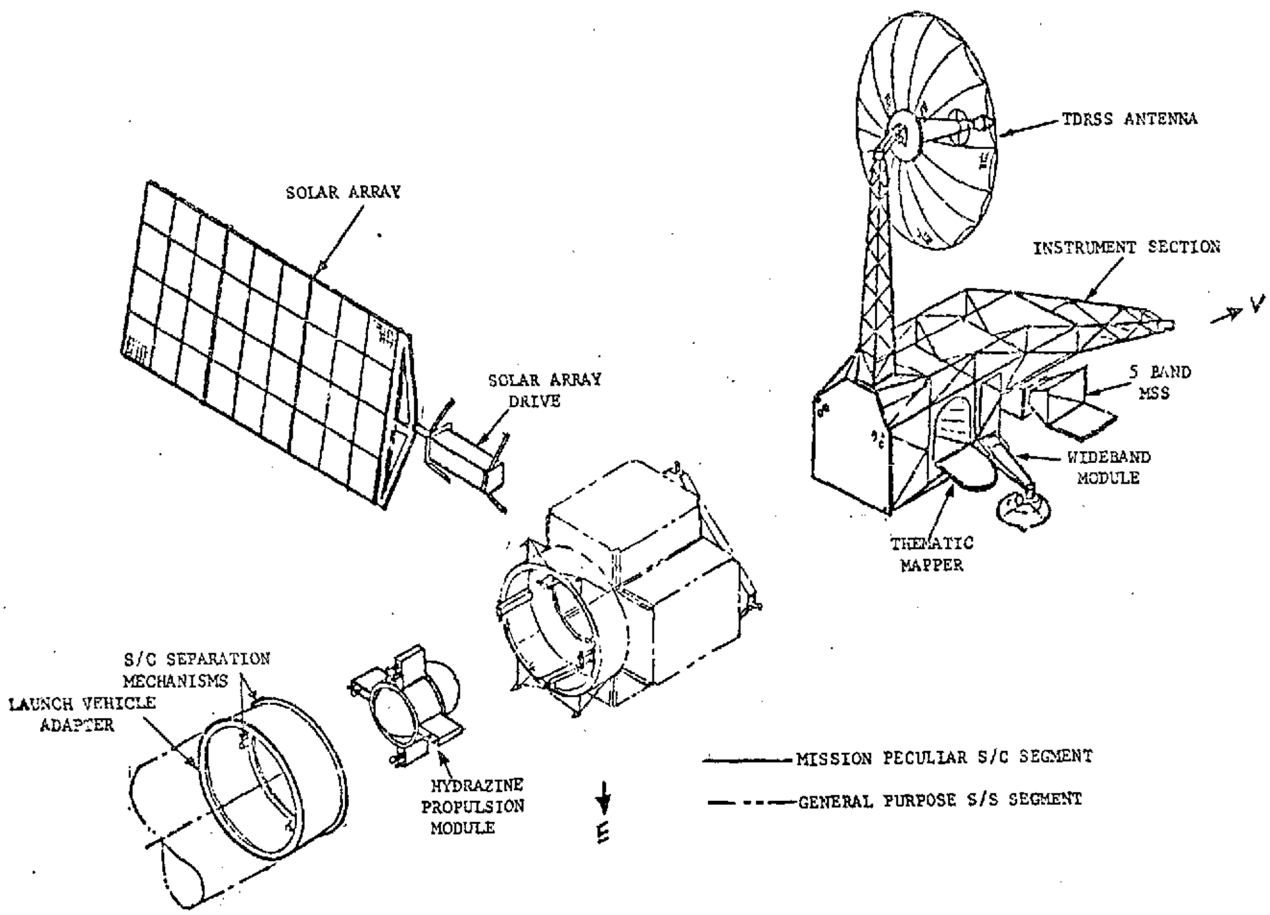


FIGURE 3.5.1.3-1 MISSION PECULIAR SPACECRAFT SEGMENT

3.5.1.4 Structures Subsystems

The Mission Peculiar Spacecraft Segment structure shall provide the primary and secondary structural elements to control the positional relationships of and provide mounting surfaces for the other subsystems. The structure shall possess sufficient strength and rigidity to maintain critical alignments of subsystems during pre-launch launch, and orbital environments.

SVS XXXX, "Specification for the EOS Structures Subsystem" defines in detail the performance and design of the structures and mechanisms required for the Mission Peculiar Spacecraft Segment.

3.5.1.5 Thermal Subsystem

The Thermal Control Subsystem is essentially a passive system which utilizes insulation, coatings, and heaters. Thermal control at the Module level will be at the following levels:

Instruments	$70^{\circ} \pm 5^{\circ}$ F
Wideband Communications	$70^{\circ} \pm 5^{\circ}$ F

The Thermal control coating will be teflon/silver.

SVS XXXX, "Specification for the EOS Thermal Control System" presents the specific details of performance and design for this system for the General Purpose Spacecraft Segment.

3.5.1.6 Wideband Communications Subsystem Module

The Wideband Communications Subsystem Module provides the means for transmitting TM and MSS Data in real time through TDRSS and also through two identical STDN S/C to ground links simultaneously. A separate link us supplied for local User Data and transmits either MSS or Compacted TM data.

* Paragraphs 3.5.1.4 through 3.5.1.11 refer to subsystem or module specifications which define in detail the requirements of each section. The referenced specifications are part of the overall Mission Peculiar Spacecraft Segment Specification and are considered to be requirements of the Mission Peculiar Spacecraft Segment.

SVS XXXX, "Specification for EOS Wideband Communications Subsystem Module" defines the details of performance and design of this module for the General Purpose Spacecraft Segment.

3.5.1.7 Mission Peculiar Software

A set of Mission Peculiar Software is required to be furnished with the on-board computer. This software will be modular so that mission unique functions may be added as required to the flight program. Basic programs are an executive which handle I/O and task management, a stored command handler; and a status buffer handler which reports observatory back-orbit status.

The above programs will be developed for the Advanced On-Board Computer (AOP). Various subsystem programs such as thermal control, power regulator control, instrument control, etc., will also be required.

SVS XXXX, "Specification for EOS Mission Peculiar Software" defines the details of the above software for the Mission Peculiar Spacecraft Segment.

3.5.1.8 Attitude Control Subsystem Module

(Described in General Purpose Spacecraft Segment Specification).

3.5.1.9 Propulsion Subsystem Module

This specification addresses only the orbit adjust, orbit maintenance, and orbit transfer functions of the propulsion subsystem.

SVS XXXX, "Specification for the EOS Propulsion Subsystem Module" defines the details of performance and design for this module for the Mission Peculiar Spacecraft Segment.

3.5.1.10 Solar Array

The Solar Array is designed to permit normal operations of the total Spacecraft and the mission peculiar payload in conjunction with the Power Subsystem Module which is treated as a General Purpose Spacecraft Segment Module.

Details of the performance and design of the Solar Array are presented in SVS XXXX, "Specification for the EOS Solar Array".

3.5.1.11 Electrical Integration Subsystem

The modular design concept of the spacecraft requires that the interface between modules be minimized and reliable. Signal distribution is primarily dependent on the command and telemetry party lines. The electrical integration of the spacecraft provides a bus capable of supporting the data flow among the ground, OBC, and spacecraft subsystems.

The requirements for electrical distribution include: (1) Signal distribution compatible with subsystem, and launch vehicle interfaces and extentof OBC control; (2) Electromagnetic compatibility requirements based on MIL-STD-461 and 462; (3) Protection against radiation environment expected for each mission; (4) Power bus protection; and (5) Command and telemetry requirements.

The design provides for two major harness segments, one on each side of the transition ring. This is consistent with the party line concept of signal distribution which minimizes the number of wires in each segment. Rack and panel, blind mate, self-aligning, deadface connectors containing signal, power, and coaxial contact are selected to simplify alignment during mating. Harnesses within each module will be segmented between subassemblies to permit easy replacement.

Details of the performance and design of the Electrical Integration Subsystem are presented in SVS XXXX, "Specification for EOS Electrical Integration Subsystem."

3.5.1.12 Scanning Spectral Radiometer (Thematic Mapper)

Details of the performance and design of the Scanning Spectral Radiometer (Thematic Mapper are defined in SVS-XXXX "Specification for the Scanning Spectral Radiometer (Thematic Mapper)".

3.5.1.13 Alignment

Critical alignment requirements for mission peculiar items are given in Table (later). The requirements are presented forinitial alignment (installation prior to environmental testing and for recheck-remeasurement after environmental test.). Also defined in the table are the measurement accuracies required. Unless otherwise specified, all alignment requirements specified are with respect to the spacecraft geometric axes.

3.5.2 Selection of Specifications and Standards

All specifications and standards other than those approved for use by NASA shall be approved by the prime contractor prior to use. Specifications and standards shall be selected in accordance with MIL-STD-143.

3.5.3 Materials, Parts and Processes

Particular attention shall be given to the application and use of materials, parts and processes to facilitate interchangeability, stocking and replacement. Materials shall be chosen on the basis of suitability and availability in the United States. Non critical materials shall be used wherever practical when performance, interchangeability or reliability will not be adversely affected or production significantly altered. Parts shall be selected from the GSFC Preferred Parts List, PPL-11. Parts, materials, and processes used in the fabrication of equipment previously accepted by the government shall be acceptable, provided that all of the following are satisfied:

a. Evidence of prior acceptance is submitted to GSFC.

b. Prior application included demonstration of capability in equivalent or more severe environments than specified in paragraph 3.4.4 herein.

c. The selection is approved by GSFC.

3.5.4 Standard and Commercial Parts

Standard or commercial parts may be used in the AGE subsystems consistent with reliability, maintainability and performance. MS or AN parts shall be used where they are suitable to the application. Commercial parts having suitable properties may be used where no appropriate standard part is available. Standard utility parts (e.g.: screws, nuts, bolts, cotter pins) may be used in the spacecraft providing they exhibit suitable properties and their use is approved by GSFC. For AGE, commercial utility parts having suitable properties may be used provided they can be replaced by standard utility parts without alteration and the corresponding standard parts numbers are referenced in the parts list.

3.5.5 Moisture and Fungus Resistance

Component design shall conform to requirement 4 of MIL-STD-454, except, for AGE, paragraph 2 and all references to MIL-STD-810 are deleted. Wherever possible, non-nutrient materials which resist damage from moisture and fungus shall be used. Protective coatings shall not be acceptable as moisture and fungus preventatives for parts which may lose their coating during the normal course of assembly, inspection, maintenance, and testing.

3.5.6 Corrosion of Metal Parts

The use of dissimilar metals, as defined in MS33586, shall be avoided wherever possible. Materials, techniques, and processes shall be selected and employed with regard to heat treatment procedure, corrosion protection, finish and assembly and installation such that sustained or residual surface tensile stress, stress concentrations, and the hazards of stress corrosion, cracking, and hydrogen embrittlement are minimized. Processes and materials for protection against corrosion of metal parts shall be selected from those specified in paragraph 3.5.3, with the exception that cadmium plating shall not be used. Selected finishes shall be compatible with all the requirements specified for each equipment. Materials and surfaces which may be exposed to an effluent shall be selected for compatibility with the effluent insofar as design considerations permit.

3.5.7 Interchangeability and Replaceability

Mechanical and electrical interchangeability shall exist between like assemblies, subassemblies, and replaceable parts, without modification. Interchangeability shall be defined in accordance with requirement 7 of MIL-STD-454.

3.5.8 Workmanship

Workmanship shall be in accordance with requirements 9 and 24 of MIL-STD-454 and with NHB 5300.4 (3A), and GSFC approved quality program plan.

3.5.9 Electromagnetic Interference

Electromagnetic compatibility shall be in accordance with the requirements of Electromagnetic Interference, Electromagnetic Compatibility (EMI/EMC) Control Plan included in Section TBD of this specification.

3.5.10 Identification and Marking

Identification and marking shall be in accordance with the requirements of MIL-STD-129, 130, MIL-E-5400, paragraph 3.1.16, and MIL-STD-1247. Shipping documentation and containers for flight equipment shall be marked "Items for Space Flight Use".

3.5.11 Electrical AGE

Electrical AGE shall be designed, fabricated and tested in accordance with the requirements of specification (later).

3.6 PERFORMANCE ASSURANCE REQUIREMENTS

3.6.1 Reliability Program

The reliability program shall be implemented in accordance with the requirements of NASA Reliability Publication NHB 5300.4 (1A), as defined in the EOS Reliability Program Plan.

3.6.2 Quality Program

The quality assurance program shall be implemented in accordance with the requirements of NASA Quality Publication, NHB 5300.4 (1B), as defined in the EOS Quality Program Plan.

3.6.3 Test Program

The test program shall be performed in accordance with the provisions of Section 4 of this specification (to be added later). Monitoring and control shall be in accordance with the provisions of the EOS Quality Program Plan.

3.6.4 Configuration Management

Configuration Control shall be maintained in accordance with the provisions of the EOS Configuration Management Plan.

3.6.5 Malfunction Reporting

Malfunctions shall be reported in accordance with the requirements defined in the EOS Failure Analysis and Reporting Plan.

3.6.6 Electrical Connections

Soldered electrical connections shall be made in accordance with the provisions of NASA Document NHB 5300.4 (3A), Requirements for Soldered Electrical Connections.

SECTION 3

3.0 SPECIFICATION FOR THE EOS STRUCTURE SUBSYSTEM

(This specification is presented in Volume 3, "Specification for the EOS General Purpose Spacecraft Segment and Modules" of Report No. 5, "System Design and Specifications".)

SECTION 4

4.0 SPECIFICATION FOR THE EOS THERMAL CONTROL SUBSYSTEM

(This specification is presented in Volume 3, "Specification for the EOS General Purpose Spacecraft Segment and Modules" of Report No. 5, "System Design and Specifications.")

SECTION 5.0

SPECIFICATION

FOR THE

EARTH OBSERVATORY SATELLITE (EOS-A)

WIDEBAND COMMUNICATIONS SUBSYSTEM

TABLE OF CONTENTS

1.0 SCOPE

This specification establishes the performance, design, interface, qualification and acceptance testing for the EOS-A Wideband Communications and Data Handling Subsystem. The subsystem is considered mission peculiar equipment specifically designed for each mission. As such the subsystem will be modularized to the extent that specific components may be readily added or deleted.

2.0 APPLICABLE DOCUMENTS

The following documents of the exact issue shown, form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and detail requirements in the following sections, the detail requirements of this specification shall supersede. In the event of conflict between documents referenced here and lower tier references to documents referenced here, the former supersede.

2.1 APPLICABLE DOCUMENTS

SPECIFICATIONS

National Aeronautics and Space Administration

EOS-410-02	Specifications for EOS System Definition Studies, 13 Sept. 1974
S-311-P-11	Quality Monitoring of Integrated Circuits, 1 June 1970
S-323-P-10	Connectors, Subminiature Electrical and Coaxial Contacts for Space Flight Use, Revised December 1969.

Military

MIL-C-38999	Connectors, Electrical, Miniature, Quick Disconnect, Est. Reliability
MIL-C-29012A	Connectors, Coaxial, RF, General Specification for
MIL-C-26482	Connectors, Electric, Circular, Miniature, Quick Disconnect
MIL-C-17	Cables, RF, Coaxial, Dual Coaxial, Twin Conductors, Twin Head
MIL-W-81044	Wire, Electric Cross-linked, Polyalkene, Insulated, Copper
MIL-E-5400K	Electronic Equipment, Airborne, General Specification for

General Electric

SVS XXXX	Specification for EOS Communications and Data Handling Subsystem Module
SVS XXXX	Specification for EOS General Purpose Spacecraft Segment
SVS XXXX	Specification for EOS Mission Peculiar Spacecraft Segment

STANDARDS

National Aeronautics and Space Administration

Aerospace Data System Standards, Part III, Associated Standards, Section I "Radio Frequency and Modulation Standard for Space-to-Ground Telemetry", prepared by GSFC Data Systems Requirements Committee, November 1965.

Aerospace Data Systems Standard, Part I, Telemetry Standards, Section 1, Pulse Code Modulation Telemetry Standard, prepared by GSFC Data Systems Requirements Committee, January 27, 1966.

OTHER PUBLICATIONS

National Aeronautics and Space Administration

NHB 5300.4 (3A) May 1968	Requirements for Soldered Electrical Connections
PPL-12 Latest Issue	GSFC Preferred Parts List
NHB 5300.4 (1A)	Reliability Program Revisions for Space Systems Contractors
NHB 5300.4 (1B)	Quality Assurance Program Provisions for Space Systems Contractor

MILITARY HANDBOOKS

MIL-HDBK-5A	Metallic Materials and Elements for Aerospace Vehicle Structure
MIL-HDBK-17	Plastics for Flight Vehicles

GENERAL ELECTRIC COMPANY

XXXXX	EOS Mission Peculiar Spacecraft Segment Quality Program Plan

XXXX Configuration Management Plan for EOS Mission Peculiar S/C Segment

XXXX Reliability Program Plan, Mission Peculiar Spacecraft Segment

3.0 REQUIREMENTS

3.1 ITEM DEFINITION

3.1.1 ITEM DESCRIPTION

The EOSA Wideband Communication Subsystem accepts, processes, and transmits data in real time from a Thematic Mapper (TM) and a Multispectral Scanner (MSS) Sensor. Four independent spacecraft-to-ground RF links are provided; a link via TDRSS which gives extra-continental coverage for TM and MSS data with a Thematic Mapper Compacted (TMC) data backup, two identical STDN links for TM and MSS with a TMC backup, and a Low Cost User (LCU) link for either MSS or TMC data. The subsystem is compatible with follow-on mission sensors and is modularized for eventual Shuttle serviceability. A block diagram of the subsystem is shown in Figure 3.1.

Four antenna subsystems are employed. The TDRSS antenna provides an S-band Transmit/Receive feed via the diplexer shown; a wideband transmit only feed at Ku band (coaxial with the S-band feed); a monopulse tracking horn complete with an electronics compliment for pointing the 8 ft. furlable reflector and a broad beam C.W. beacon source (shown separate but actually fed into the monopulse sum port via a diplexer). The monopulse tracking loop is closed thru a D.C. torqued AZ-EL gimbal drive with its associated electronics and rotary R.F. joints. An override is provided to enable course pointing via command from the OBC located in the C&DH module.

The second two antennas for the STDN link are identical parabolic dishes with X-band feeds for transmit only. The antennas are independently open-loop pointed by means of stepper motor driven X-Y gimbals on command by the OBC.

The fourth antenna, for the LCU link, is a fixed, shaped beam design which illuminates a ground track $\pm$ 500 Km from nadir. A feed at X-band is supplied.

The frequency allocations and spectral content of each antenna is as shown in Figue 3.1.

The TDRSS antenna subsystem is mounted on a deployable boom above to spacecraft. The STDN and LCU antennas are mounted on the earth side of the spacecraft.

The antennas are serviced by a complement of TWT power amplifiers. All TWTA's are complete with output bandpass filters, output isolators and H.V. power supplies. The CW Beacon Generator and TM QPSK signals are summed and amplified by 13 watt KU band unit. This signal is R.F. Mux'ed with the PCM/FM signal amplified by a 2 W KU band unit. The MSS PCM/FM signal may be replaced by the TMC PCM/FM signal on command and at the same R.F. power level. The Mux'ed signal spectrum feeds the TDRSS antenna thru a single R.F. rotary joint. A DEMUX strips off the beacon.

The STDN antennas are simultaneously fed from the summed outputs of a 3.3W TM QPSK signal and a 0.7 W MSS PCM/FM signal. The MSS PCM/FM signal may be replaced by the TMC PCM/FM signal on command and at the same R.F. Power level.

The LCU antenna is driven directly by a 30 W TWT which amplifies either the MSS PCM/FM signal spectrum or the TMC PCM/FM signal spectrum.

The QPSK modulator accepts the serial NRZL binary data signal at 15 MBS and simultaneously provides a modulation spectrum at Ku and X band. The Two PCM/FM modulators, shown separately, is a single unit with an input switching network to select MSS and TMC data. The outputs are as shown. The PCM/FM converter is a separate unit which supplies the required PCM/FM modulation D.C. voltages. The QPSK modulator power converter is self contained.

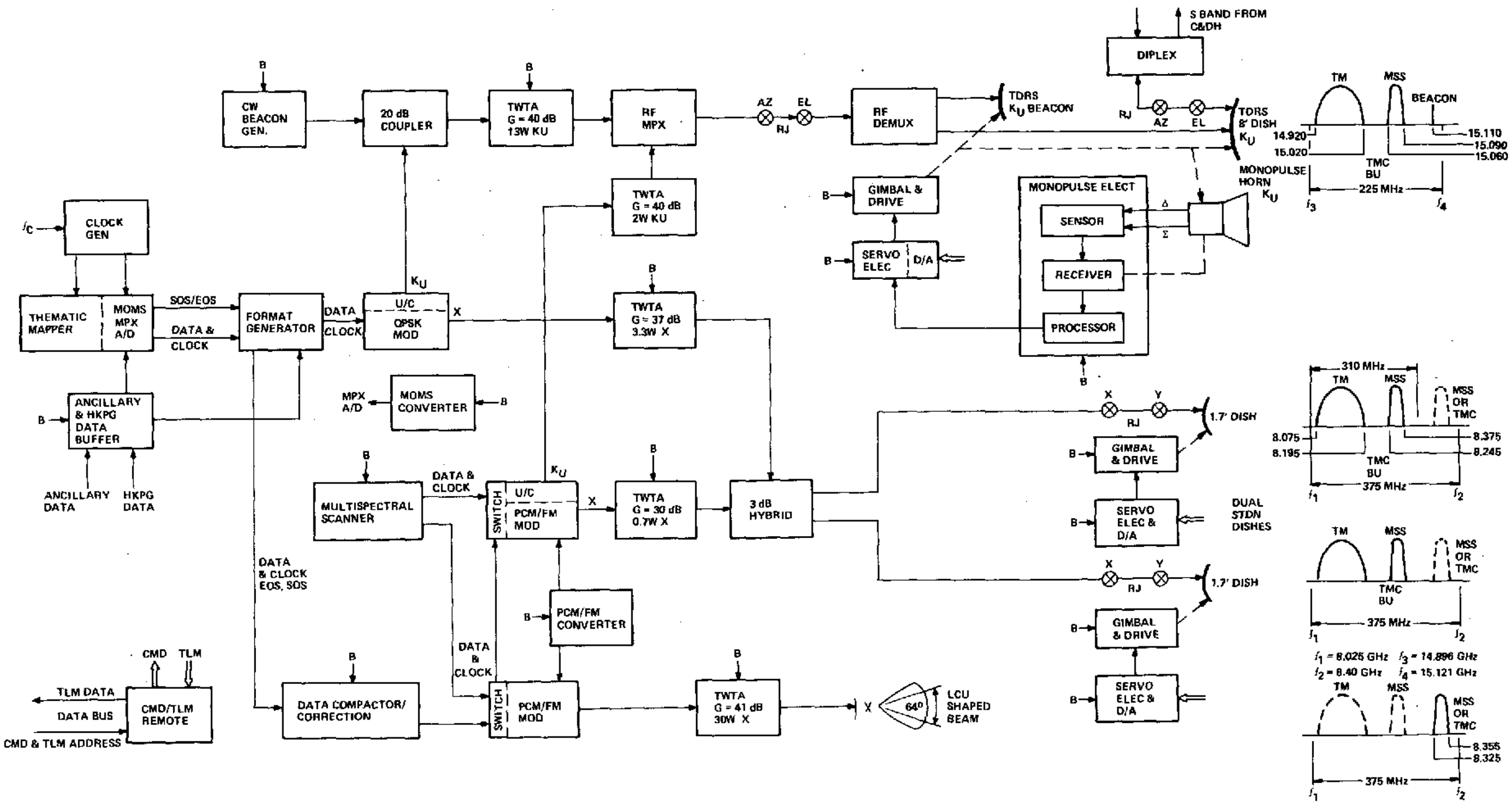


Figure 3-1. Wideband Subsystem

The MOMS multiplexer/A-D converter is integrally mounted with the TM sensor and serves to sequentially sample and quantize the multiplicity of sensor analog voltages and HKPG data. The MOMS power converter supplies the necessary D.C. operating voltages. Nominal output bit rate is 67 MBS.

The Format Generator provides a preamble, a minor frame sync and integrates ancillary data into the digital bit stream.

The compactor/corrector accepts formatted data and provides the memory, logic and processing circuitry to compact TM data according to 4 commandable options to a bit rate of 15 MBS. In addition, the compacted data is "x" geometrically corrected.

A remote command/telemetry unit provides for command decoding and implementation and provides telemetry data to the C&DH module.

The wideband subsystem is clocked in coherence with the spacecraft clock.

All wideband conponents except the MOMS PBX/AR and antennas are housed in a 38" x 34" x 12" module.

3.1.2 INTERFACE DEFINITION

3.1.2.1 Electrical

The electrical interface with the wideband module is via seven separate connectors mounted on the module surface. Four coaxial R.F. connectors provide the interface with the four antenna subsystems. A fifth connector provides power, command, telemetry, clock time code and ancillary data. The signal distribution on this connector is shown in Table 3.1. A sixth connector provides a test interface for checkout of module functions using GSE.

The MOMS/TM interface is a special multipin connector capable of interfacing 84 single ended, shielded cable, analog inputs.

Table 3.1. Module Electrical Interface

Signal	# Pins	Cable
Input Power	2 + 2 RTN	T2
Heater Power	1 + 1 RTN	T2
Supervisory Data Bus	2 + 2 RTN	T2S
Return Data Bus	2 + 2 RTN	T2S
Module Signal Return	2	SC
Shield Tie (Chassis Gnd)	10	---
1.6 MHz Clock	1 + 1 RTN	Twin-ax
Timecode	2 + 2 RTN	T2S
Ancillary Data	2 + 2 RTN	T2S

3.1.2.2 Mechanical

The wideband module is mounted directly to the spacecraft structure.

3.1.2.3 Thermal

The wideband module is thermally isolated from the spacecraft structure. Heat from the module is radiated outboard from the module sides and bottom. Thermal control is via module heaters under OBC control.

3.2 CHARACTERISTICS

3.2.1 PERFORMANCE

3.2.1.1 Operating Modes

Operating modes will be as follows:

a. TM and/or MSS with Tracking Beacon via TDRSS. TMC to backup MSS on command.

b. TM and/or MSS via the dual STDN links. TMC to backup MSS data, on command.

c. TMC or MSS via the LCU link based on real time or delayed command.

d. A radiometric calibration mode providing the capability for storing sensor minor frames in OBC storage during the calibration cycle.

e. A TDRSS acquisition mode for the following sequence of events:

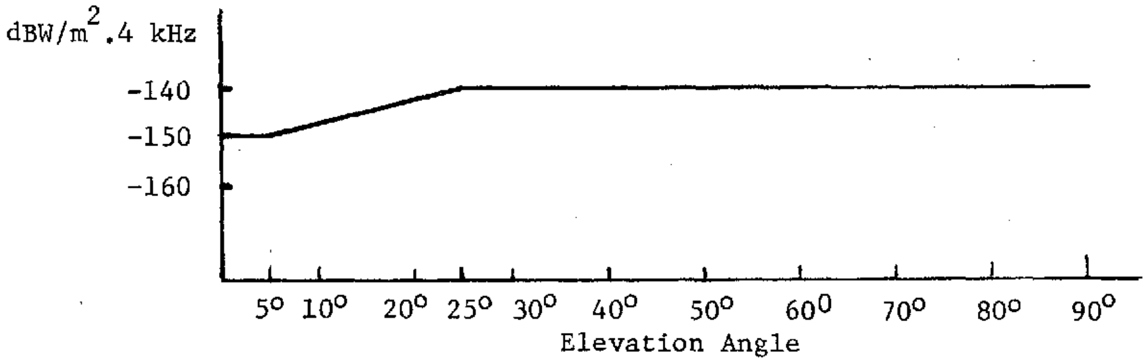
1. Ground station command via TDRSS to enable KU band beacon.
2. Slew. EOS dish to illuminate TDRSS and enable beacon.
3. TDRSS acquires EOS beacon and autotracks EOS.
4. TDRSS transmits 10 dBW EIRP CW beacon.
5. Ground station command via TDRSS to acquire TDRSS beacon.
6. EOS acquires and autotracks TDRSS beacon.
7. Ground station command via TDRSS to transmit data.
8. Wideband transmission enabled.

3.2.1.2 RF Requirements

a. Frequency Allocation

EOS/TDRSS	14.896 - 15.121 GHz	
STDN + LCU	8.025 - 8.40 GHz	
TDRSS Beacon	13.75 - 13.8 GHz	

b. Power Flux Density (PFD). The incident PFD at X-band shall not exceed the following limits



Transmission via TDRSS will conform to the following formula

EIRP (dBW) $\leq$ Data Rate (dB) - 25.1

c. Bit Error Rate (BER). The hardware performance parameters EIRP, link margins, and interlink crosstalk will give a BER $\leq 10^{-5}$ for all links under worst case conditions.

d. Spurious Frequencies. Frequencies outside the allocated bandwidths shall be at least 10 dB below the unmodulated carrier level at the band edge and roll off at 18 dB/octave to a level $\geq$ 70 dB below the unmodulated carrier.

e. Antenna Coverage

STDN. Both dishes pointable over $\pm$ 60° from nadir about two axes. Pointing precision $\pm$ 0.6°.

TDRSS. Dish pointable over $\pm$ 120° about two axes. Open loop precision $\pm$ 0.6°. Monopulse tracking to $\pm$ 0.2°.KU band beacon at + 30 dBW.

LCU. Shaped beam. Beamwidth is $\pm$ 32° from nadir.

f. Detailed QPSK Performance Parameters. The following parameters shall be maintained over the specified operating voltage range, life and environmental temperature range of 20 $\pm$ 10°C.

Modulator Phase Unbalance	$\pm$ 2.5 degrees
Modulator Amplitude Unbalance	$\pm$ 3%
Modulator Rise Time	10% of the symbol period
Subsystem AM/PM conversion factor	6 degrees/db
Bandwidth Limiting and Data Detector Mismatch	100 MHz minimum
Amplitude Variation over $\pm$ 50 MHz around the carrier	1.5 db p-p ripple
Subsystem Phase Nonlinearity	Parabolic - 15 deg. Cubic - 15 deg. Ripple - 15 deg.
Modulator Data Assymmetry	1.1
Data Synchronization	Skewed 0.5 bit $\pm$ 0.25 bit

g. Detailed PCM-FSK Link Performance Parameters. The following parameters shall be maintained over the specified operating voltage range, life, and environmental temperature range of 20 $\pm$ 10°C.

Data Symmetry. 1.05

Peak Deviation. The modulator shall be capable of a peak deviation at least $\pm$ 6 MHz at modulating frequencies from D.C. to 20 MHz.

Modulation Sensitivity. The W.B. Modulator shall be set for a peak frequency deviation of $\pm$ 5.6 MHz about the carrier center frequency at the modulator output, with a simulated data and clock signal at each of the data and clock input lines. The peak deviation will vary no more than $\pm$ 6% (5.95 MHz max. and 5.26 MHz min.) over the operating temperature and voltage variation limits. A frequency increase shall result from a positive going voltage on the RBV inputs, and from a data "0" at the input.

Sensitivity Variation. The deviation sensitivity shall not vary more than $^{+0.2}_{-0.8}$ db from the 100 KHz value over the frequency range from dc to 3.2 MHz or $^{+0.2}_{-3.7}$ db from dc to 10 MHz. This includes the effects of the pre-modulation Bessel filters.

Modulation Linearity. The output frequency deviation shall not vary more than 120 KHz from a straight line fitted between the ± 6 MHz deviation points on the deviation vs. input dc voltage curve.

Intermodulation Products. With a single sinusoid applied to the W.B. Modulator causing a ± 5.6 MHz peak deviation, the amplitude of the 2nd harmonic distortion measured in a ground receiver output shall be no greater than -25 db with respect to the fundamental at the receiver output, when the input frequency is varied between 100 KHz and 3.5 MHz. The 3rd harmonic under the same conditions shall be no more than -27 dB with respect to the fundamental.

Incidental Frequency Modulation. Incidental rms frequency modulation shall be at least 50 dB below the peak deviation signal level.

Signal to Interference Ratio. The demodulated video output signal to interference ratio shall exceed 50 dB from dc to 800 KHz and shall decrease linearly between 800 KHz and 3.2 MHz to 35 dB minimum. Interference sources may include S/C generated signals or signals generated within the WBCSS.

Linear Group Delay. The total S/S linear component of Group delay across either 20 MHz operating band shall not exceed 0.55 nanosec/MHz.

Parabolic Group Delay. The total S/S parabolic component of Group delay across either 20 MHz operating band shall not exceed 0.08 nanosec/MHz.^{2}

Baseband Group Delay Variation. Group delay variation shall not exceed ± 8 nanoseconds from 100 KHz to 3.5 MHz or ± 15 nanoseconds from 100 KHz to 10 MHz. Group delay variation from dc to 100 KHz shall not exceed TBD nanoseconds.

R.F. Amplitude Response. The peak R.F. amplitude versus frequency response shall vary no more than 1 dB over either of the 20 MHz R.F. operating bands.

3.2.1.3 Compaction

Quantized TM data at 67 MBS will be compacted per the commandable options shown in Table 3.2.

TABLE 3.2 Compaction Modes

	Ground Resolution (Meters)	Spectral Bands Used	Swath Width	Data Rate After Compaction, Correction and Formatting (MBS)
#1	60 x 60	All 6	Full	15
#2	30 x 30	Any 2 of the first 5 bands + Band 6	1/2	15
#3	30 x 30	All 6	1/4	15
#4	30 x 30	Any 1 of the first 5 + Band 6	Full	15

3.2.1.4 Sensor Multiplexing/Quantisizing

The TM channels must be sequentially sampled, quantized, formatted, and interleaved with ancillary and telemetry data in a manner which facilitates recovery at the ground station. The baseline design is based on the following parameters:

Analog Channels	84
Thruput bit rate	67×10^6 BPS
Quantization level	7 bits
Swath time	71 msec
Scan efficiency	80% max.
Analog degradation	$\pm$ 1/2 bit
Aperture ambiguity	50 pico seconds

3.2.2 DESIGN

3.2.2.1 Electrical

3.2.2.1.1 Power

The wideband module shall be capable of meeting specification when operating from a regulated bus voltage of +28 $\pm$ 0.3 VDC with a ripple not to exceed 100 mv p-p. The bus source impedance will not exceed 0.1 ohms from 0 to 10 KHz. The wideband module shall survive power subsystem failure modes where the bus voltage varies from +18 to +33 volts and a volt second product not to exceed 250 u volt seconds. The wideband module shall not present a step change greater than 2 volts with a volt product not exceeding 100 u volt seconds. The power dissipation shall not exceed 550 watts peak load.

3.2.2.1.2 Command

Command control of the components in the wideband module shall be obtained via two remote decoder/TLM muxes located in the module. The command interface is as defined in Paragraph 3.2.1.2 of the "Specification for the EOS C&DH Subsystem Module". Relay driven circuits shall be provided within the components as necessary.

3.2.2.1.3 Telemetry

Telemetry data acquisition for each wideband module component shall be obtained via two remote decoder/TLM muxes located within the module. The telemetry inputs are as defined in Paragraph 3.2.1.2 of the "Specification for the EOS C&DH Subsystem Module".

3.2.2.1.4 Outputs/Inputs

The wideband module shall provide four R.F. coaxial interfaces. One interface shall be with the TDRS antenna, two with the dual STDN antennas and the fourth with the LCU antenna. The supervisory and return data busses and the timecode data bus shall be provided as redundant T25 cable outputs with one of each bus energized (via command) at any one time. The inputs from the TM & MSS data lines shall be coaxial and differential shielded lines with separate lines for data and clock.

3.2.2.1.5 Grounding

Three separate grounding busses shall be employed; a power ground bus, signal ground bus and a module chassis bus.

3.2.2.1.6 Redundancy

There shall be no component redundancy included in the wideband module.

3.2.2.1.7 Electromagnetic Compatibility

The wideband module shall employ good design practices in chassis design, shielding, EMC filtering, grounding, bounding, etc so as to minimize radiation of self generated noise and susceptability to EMI from external sources.

3.2.2.1.8 Harness

The module harness shall provide all electrical interfaces between subsystem assemblies within the module and to the module/structure interface and test connectors. The harness shall be of modular design for maximum system flixibility. Installation or removal of the harness should be possible without removing electrical assemblies. It shall be possible to remove

electrical assemblies without removing the harness. Cable strain relief or backshell potting shall be employed at all harness terminations.

Wire sizes shall be selected to hold round trip voltage drops between source and load to one percent or less of the supply voltage. The minimum wire size for power and control circuitry shall be AWG #20. The minimum wire size for data or test circuitry shall be AWG #22. Under the worst case conditions, wire temperature shall not exceed the temperature rating of the wire insulation.

3.2.2.2 Mechanical

The wideband module shall be housed in a container having the dimensions 38" x 34" x 12". The internal assemblies shall have simple bolt mounting and electrical connector interfaces. The weight of the C&DH module shall not exceed 215 pounds.

SECTION 6

SPECIFICATION

FOR THE

EARTH OBSERVATORY SATELLITE (EOS-A)

MISSION PECULIAR SOFTWARE

Table of Contents

SECTION 1

SCOPE

This specification establishes the performance and interface requirements for the applications packages necessary for operation of the EOS-A. Software application packages are included for the antenna pointing (STDN + TDRSS), payload (TM), and shuttle functions. Each of these packages defines the interfaces and computational requirements associated with the function and includes a flow diagram showing the basic approach to the software package.

SECTION 2

APPLICABLE DOCUMENTS

TBD

SECTION 3.0

REQUIREMENTS

3.1 ANTENNA POINTING APPLICATION PACKAGE

The Antenna Pointing application package (AP) shall control the gimballed motion of three EOS-A antennas. The specific AP functions are:

(1) Monitor the transmitting/receiving initialization and termination commands issued via the Command Storage and Sequencing application package (CSS).

(2) When communications involving any one or more of the three antennas is in progress, calculate the pointing updates for these antennas.

(3) Generate commands (to be issued by the CSS) reflecting either initial pointing positions of the antennas or the calculated pointing updates.

Two of the three antennas are pointed either at the STDN stations or at the LCU stations. Transmission from each of these antennas occurs at intervals designated by prestored initialization and termination commands (which are executed, in accordance with their time tags, by the CSS). Each related initialization/termination command pair - which cover an average transmission period of a few minutes - shall have an associated initial pointing position (pitch and roll angle) supplied with it. When the (AOP) Executive detects that a transmission initialization command (with an associated initial pointing position command) has been executed, the Executive will subsequently transfer control to the AP at intervals occurring somewhat in advance of when the next pointing update command should be issued. Updates shall occur at a maximum rate of ten times per second. The AP shall use the latest ephermis data in conjunction with the initial pointing position to compute the first pointing update. Subsequent updates employ the latest ephemeris data in conjunction with the last calculated pointing position. Stored algorithms are used as the basis for update calculations. The STDN/LCU pointing requirement is in 1.5

degree increments over a 120 degree range in each axis. The AP shall transfer a command(s) reflecting the latest update to the CSS via the Executive.

The third antenna is pointed at the TDRSS and requires open loop pointing information from the OBC when this antenna is transmitting and/or receiving at S-Band or for initial acquisition of the monopulse carrier for transmission at Ku band. The TDRSS pointing requirement is also in 1.5 degree increments, but must encompass a range of 240 degrees in each axis. The procedure for handling the TDRSS antenna updates is similar to that described for the antennas pointed at the STDN or LCU stations. However, the algorithms employed for the TDRSS antenna pointing update calculations also require and incorporate ephemeris data for the TDRSS spacecraft. Figure 3.1-1 is a flow chart of the Antenna Pointing Application Package.

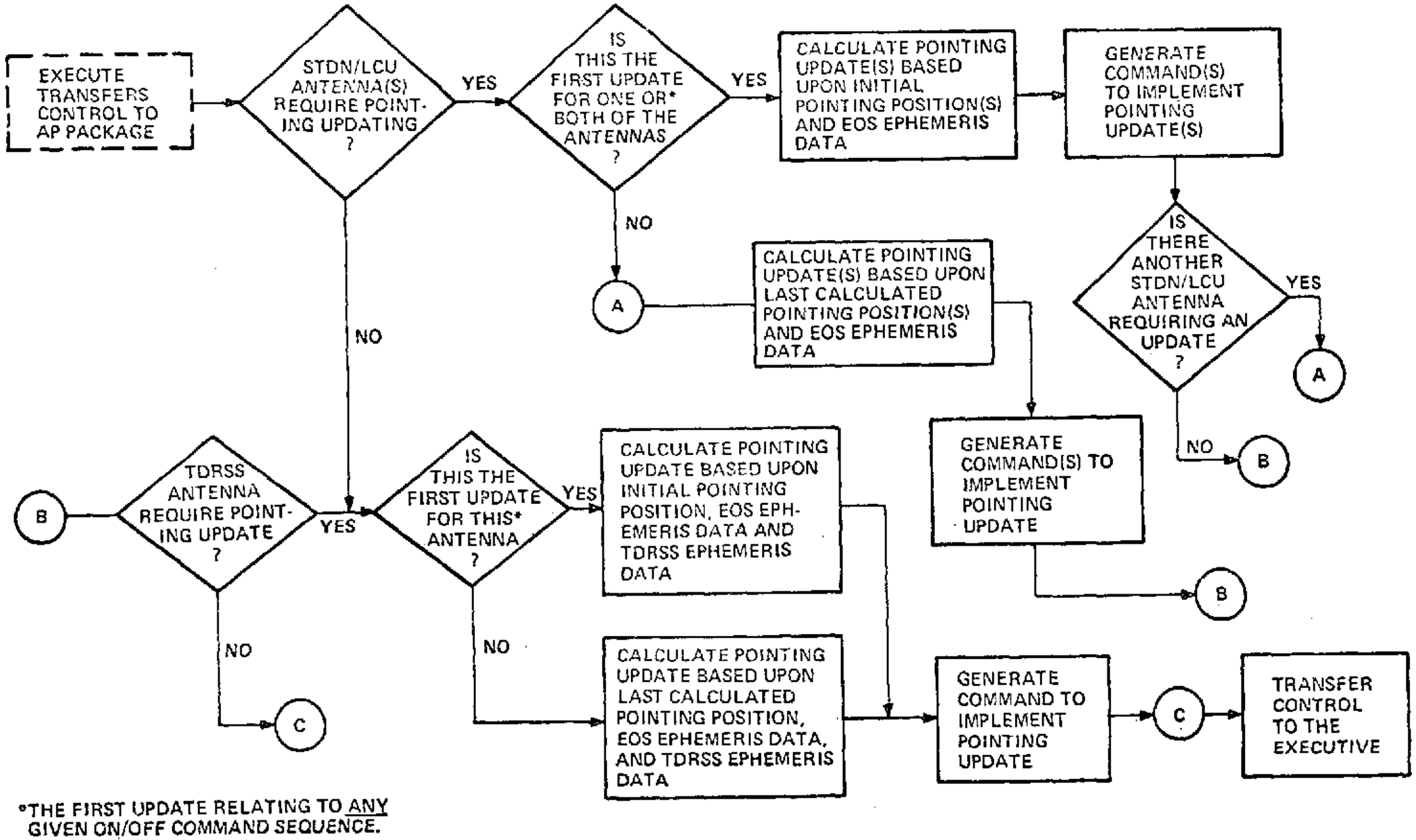


Figure 3.1-1. Flow Chart of Antenna Pointing Application Package

3.2 PAYLOAD APPLICATION PACKAGE

Many of the OBC functions to be performed in support of the Payload Application Package (PAP) shall be performed by a) the Telemetry Data Handling and b) the Command Storage and Sequencing Application Package Packages. These functions shall include:

1) Issuance of (ground-originated) turn on/ turn off commands to the payload (PL) instruments.

2) Issuance of (ground-originated) commands to position the instrument(s) and perform aperture correction.

3) Performance of limit, status, and alarm checking.

4) Issuance of commands (To MOMS) to modify the video data being transmitted via MOMS to small ground stations.

5) Generation of ancillary data for insertion into wideband module data stream .

The PAP should generate ancillary data for insertion into the video stream via the wideband module. This data shall comprise ephemeris, time correction (update), attitude, attitude rate, alignment bias, and sun calibration data. The ephemeris data, which is periodically updated from the ground on an approximately once per orbit basis and frequently algorithmically updated by the ACS, is transmitted to the wideband module once per second along with attitude and attitude update data (also generated by the ACS Application Package. The time correction, alignment bias, and sun calibration data are updated about once per orbit, and transmitted to the wideband module once per minute. The PAP shall assemble the auxilliary data into a buffer area, and generate a control word, to be interpreted by the Executive, designating when auxiliary data is to be injected into the TLM and the buffer area in which this data is located.

Figure 3.2-1 is a flow chart of the Payload Application Package.

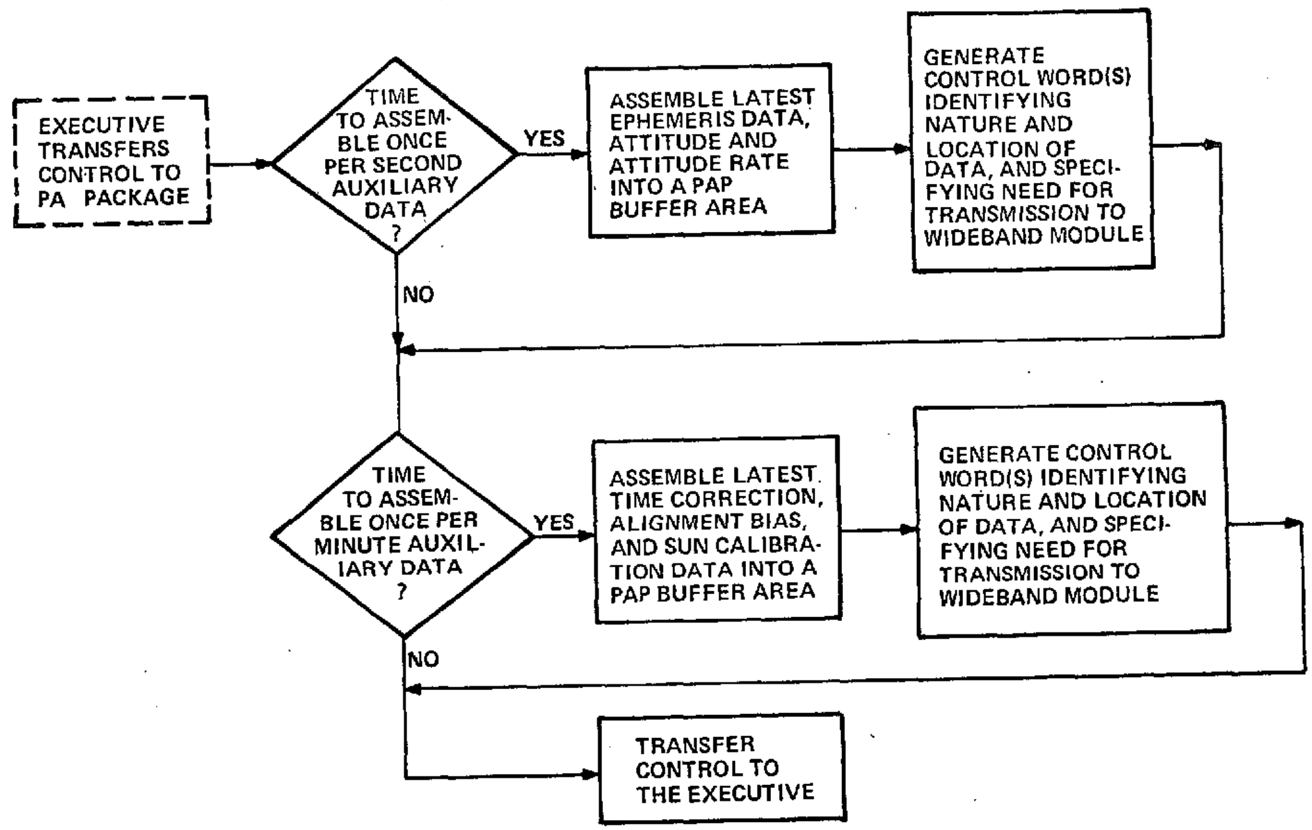


Figure 3.2-1. Payload Application Package

3.3 SHUTTLE APPLICATION PACKAGE

The Shuttle Application Package (SP) shall perform the following functions:

1. Validating the 128 bit commands transmitted by the Shuttle to the EOS (via an OBC DMA input channel tied to the EOS/Shuttle.

2. Extracting from the Shuttle command a 32 bit "command data" field. A sequence of such 32 bit fields shall be used to comprise 40 bit commands analogies to that received from the ground by the Central Command Decoder (CCD)

3. Validating and interpreting the 40 bit command in a fashion analogous to that performed by the CCD.

4. If the "operations code" field of the 40 bit command signifies immediate command issuance, reformatting the 40 bit command into a 32 bit command party line format and initiating transfer of this command (via the Executive) to the Telemetry Format Generator.

5. If the "operations code" field of the 40 bit command designates a delayed command or computer load command, extracting and storing the appropriate 24 bit "sub-command" field for subsequent processing by the Command Storage and Sequencing Application Package (CSS).

6. Generating (including formatting) telemetry data reflecting key EOS subsystem /payload parameters words for transmission to the Shuttle via an OBC DMA output channel tied to the EOS/Shuttle umbilical.

The validation of the 128 bit shuttle command by the SP consists of checking the orbiter and EOS address codes to assure conformance with this particular umbilical link.

The extraction of the 32 bit command data field by the SP is accomplished by recognition of the immediately preceding unique synch pattern sequential 32 bit command data fields are essentially concentrated by the SP to constitute 40 bit commands analogous to those received by the CCD.

The validation of the 40 bit command by the SP consists of checking the 7 bit S/C address (to insure conformance with the EOS preassigned address), and performing a polynomial check utilizing the 7 bit polynomial check code field.

Figure 3.3-1 is a flow chart of the Shuttle Application Package.

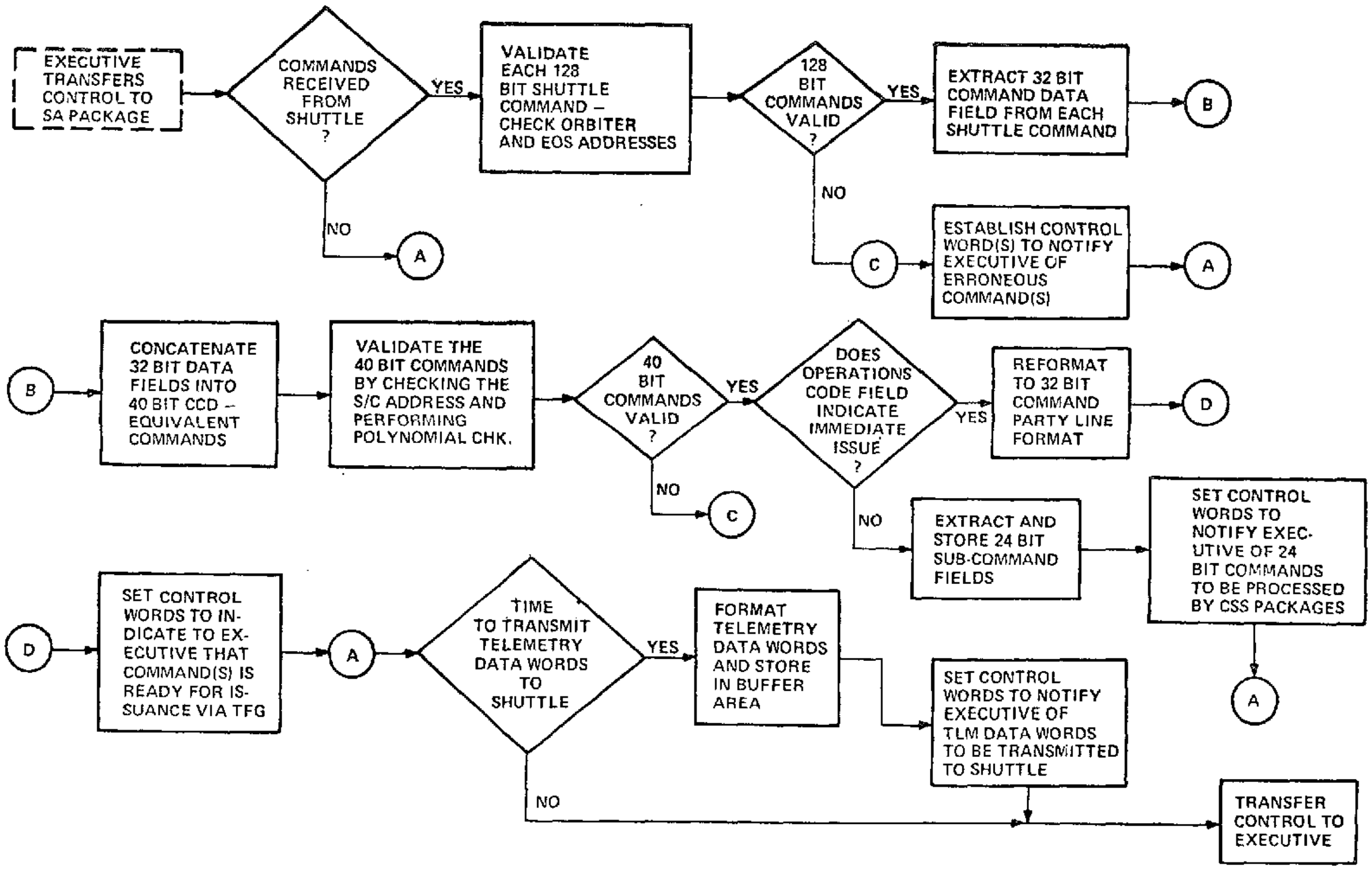


Figure 3.3-1. Flow Chart of the Shuttle Application Package

Fig. 3.3-1

SECTION 7.0

SPECIFICATION SVS-XXXX

16 SEPTEMBER 1974

SPECIFICATION FOR

THE

EOS HYDRAZINE PROPULSION SUBSYSTEM MODULE

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

1.0 SCOPE

This specification establishes the performance, design, and test requirements for a Hydrazine Propulsion Subsystem, hereinafter referred to as the PS. The PS is a monopropellant (hydrazine) fueled, varying thrust (blowdown) rocket engine system which is used to perform various orbit transfer, orbit adjust and attitude control functions for an earth orbiting spacecraft.

2.0 APPLICABLE DOCUMENTS

The following documents (of the exact issue shown) form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and the detail requirements of Sections 3, 4, and 5, the detail requirements of Section 3, 4, and 5 shall supersede. In the event of conflict between documents referenced here and lower tier references in documents referenced here, the former shall supersede.

Military	
MIL-P-26536C	Propellant Hydrazine
MIL-P-27401B	Propellant Presurizing Agent, Nitrogen
MIL-A-18455B	Argon, Technical
MIL-P-27407 Supple. I	Propellant, Helium, Pressurizing
MIL-STD-454C	Standard General REquirements for Electronic Equipment
Federal	
TT-I-735	Isopropyl Alcohol
NASA	
NHB 5300.4 (3A) May 1968	Soldering of Electrical Connectors
Air Force	
AFWTRM-127-1	Air Force Western Test Range Safety Manual

General Electric

TBD	Electromagnetic Compatibility Requirements for Components and Subsystems
TBD	Environmental Design and Test Requirements for Components and Subsystems
TBD	Approved Materials and Processes List
TBD	Electrical System Interface Requirements
TBD	Harness Design Requirements
TBD	Approved Parts List

Drawings

General Electric

TBD	Propulsion Module Envelope Drawing
TBD	Propulsion Module Structural Assembly

3.0 REQUIREMENTS

3.1 ITEM DEFINITION

The PS is a monopropellant hydrazine type propulsion system of single module construction consisting of a propellant storage and expulsion section, a propellant distribution section, and a rocket engine section. The PS provides the mass expulsion used for performing the following functions:

(a) Orbit Transfer Functions

 (1) Impart Large Velocity Change Impulses Required for the Establishment of the Spacecraft Retrieval Orbit

(b) Orbit Adjust Functions

 (1) Removal of Launch Vehicle Induced Injection Orbit Velocity Errors

 (2) Maintenance of Orbital Parameters

 (3) Spacecraft Attitude Control During Orbit Transfer Firings

(c) Attitude Control Functions

 (1) Initial Stabilization of the Spacecraft

 (2) Reaction Torque to Counteract the Torque Produced during Momentum Wheel Unloading

(3) Restabilization of the Spacecraft to the Celestial References

(4) Spacecraft Disturbance Torque Removal which are Generated during Orbit Adjust Firings

(5) Limit Cycle Attitude Control of the Spacecraft

3.1.1 ITEM DIAGRAMS

The PS block diagram defining the flow schematic and component location is shown in Figure 1. The PS shall be designed as a single module of all welded or brazed construction (upstream of all engine valve seats) and shall be capable of spacecraft installation as a completely assembled subsystem (without propellant and pressurant) on the PS propulsion module structure.

3.1.2 INTERFACE DEFINITION

3.1.2.1 Power

All electrically operated components of the PS shall operate from a power source having characteristics as defined in Paragraph 3.0 of General Electric Document (TBD).

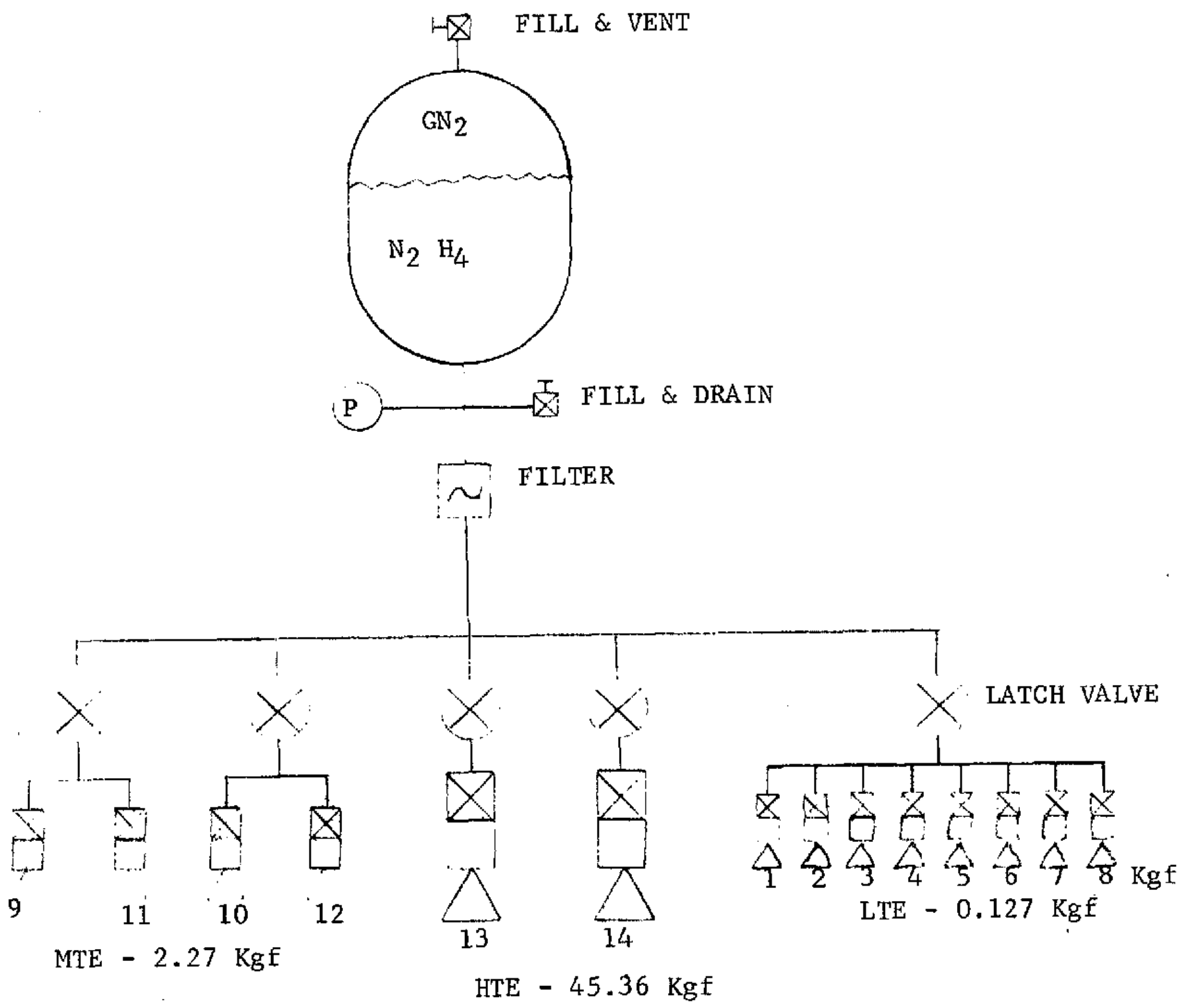
3.1.2.2 Telemetry

The PS shall provide for telemetry measurement points at locations identified in Table 1. Telemetry measurements shall have interface characteristics as defined in Paragraph 5.0 of General Electric Document (TBD).

Pressure sensor calibration data provided by the subcontractor shall be a minimum of five (5) data points plotted in degrees centigrade versus volts. The pressure sensor calibration data shall have an accuracy of $\pm 1\%$ of the full scale measurement range. The temperature sensor calibration data shall have an accuracy of $\pm 55^{\circ}C$ throughout a measurement range of $0^{\circ}C$ to $260^{\circ}C$ and shall have a measurement capability of from $-29^{\circ}C$ to $955^{\circ}C$.

FIGURE 1

PROPULSION SUBSYSTEM BLOCK DIAGRAM



FUNCTION	LTE	MTE	HTE
+ Roll	1 and 5 or 3 and 7		
- Roll	2 and 6 or 4 and 8		
+ Pitch	3 and 8	11	
- Pitch	4 and 7	9	
+ Yaw	2 and 5	12	
- Yaw	1 and 6	10	
Orbit Adjust		9 and 11 or 10 and 12	
Orbit Transfer			13 or 14

3.1.2.3 Thermal Control

The PS, exclusive of the rocket engines, shall be thermally controlled during orbit to the limits specified in Paragraph 3.1 of GE Specification (TBD).

The PS shall incorporate the required rocket engine electrical heaters in order to meet the performance requirements of Paragraph 3.2.1 under the environmental conditions of deep space. These conditions shall include the seasonal variation of the solar flux as well as nominal tolerances as follows:

Solar Flux $S_o = 1353 \pm 13.5$ Watts/m^2

Seasonal Variation = +46.4, -44.1 Watts/m^2

Rocket engine electrical heaters shall be controlled by Ground Command.

3.1.2.4 Structure

The PS shall be assembled on a structure shown on GE drawing (TBD).

The PS module space envelope and rocket engine orientation shall be as defined on GE Drawing (TBD). The PS module shall withstand the static and dynamic environments of Paragraph 3.1 of GE Specification (TBD).

Table 1
Propulsion Subsystem
Telemetry Measurement Points

Propellant Feed Pressure

Propellant Tank Temperature

REA Chamber Temperature (1)
REA Chamber Temperature (2)
REA Chamber Temperature (3)
REA Chamber Temperature (4)
REA Chamber Temperature (5)
REA Chamber Temperature (6)
REA Chamber Temperature (7)
REA Chamber Temperature (8)
REA Chamber Temperature (9)
REA Chamber Temperature (10)
REA Chamber Temperature (11)
REA Chamber Temperature (12)
REA Chamber Temperature (13)
REA Chamber Temperature (14)

Latch Valve Position (1)
Latch Valve Position (2)
Latch Valve Position (3)
Latch Valve Position (4)
Latch Valve Position (5)

3.1.2.5 Aerospace Ground Equipment

The PS shall provide for interfaces with the following items of AGE:

(1) Propellant and Pressurant Servicing Cart

(2) Rocket Engine Nozzle Alignment Targets

(3) Subsystem Electrical Test Set

(4) Subsystem Shipping Container

3.1.3 MAJOR COMPONENT LIST

The PS is composed of the following components:

(1) Low Thrust Engine (8)

(2) Medium Thrust Engine (4)

(3) High Thrust Engine (2)

(4) Propellant Tank Assembly (1)

(5) Fill and Drain Valves (2)

(6) Latching Valves (5)

(7) Propellant Filter (1)

(8) Pressure Transducer (1)

(9) Electrical Harness (1)

(10) Support Structure (1)

3.2 CHARACTERISTICS

3.2.1 PERFORMANCE

3.2.1.1 High Thrust Engines

The PS High Thrust Engines (HTE) shall have the following performance characteristics:

3.2.1.1.1 Thrust Level

Each HTE shall provide an initial thrust of 45.36 $\pm$ 5% Kgf. This thrust level shall be achieved under initial tank pressure conditions at vacuum using 20°C propellant and pressurant. The HTE shall be capable of operation over a thrust ratio (initial thrust to final thrust) of 3.0.

3.2.1.1.2 Total Impulse

Each HTE shall be capable of providing 9285 Kgf-sec of total impulse.

3.2.1.1.3 Duty Cycle

In any single operation at initial tank pressure conditions, each HTE shall be capable of a minimum impulse burn of 25 Kgf-sec and a maximum impulse burn of 5000 Kgf-sec. Each HTE shall be capable of 50 on-off cycles of which all shall be initial starts. The HTE's shall not be duty cycle limited.

3.2.1.1.4 Predictability

The impulse predictability for each HTE shall be $\pm$6% for total impulse burns in excess of 225 Kgf-sec.

3.2.1.1.5 Specific Impulse

The steady state minus three sigma specific impulse for a HTE steady state burn at initial thrust level conditions and with the propellant at 20°C shall exceed 230 Kgf-sec/Kgm.

3.2.1.2 Medium Thrust Engines

The PS Medium Thrust Engines (MTE) shall have the following performance characteristics.

3.2.1.2.1 Thrust Level

Each MTE shall provide an initial thrust of 2.27 $\pm$ 5% Kgf. This thrust level shall be achieved under initial tank pressure conditions at vacuum using 20^{o}C propellant and pressurant. The MTE's shall be capable of operation over a thrust ratio (initial thrust to final thrust) of 3.0.

3.2.1.2.2 Total Impulse

The PS shall provide a minimum total impulse of 3260 Kgf-sec for accomplishing the spacecraft MTE functions. Any single MTE shall have the capability of providing 2500 Kgf-sec of total impulse.

3.2.1.2.3 Duty Cycle

In any single operation at initial tank pressure condition, each MTE shall be capable of a minimum impulse burn of 0.11 Kgf-sec and a maximum impulse burn of 450 Kgf-sec. Each MTE shall be capable of 25,000 on-off cycles. Of these, 5000 cycles shall be initial starts. The MTE's shall not be duty cycle limited.

3.2.1.2.4 Predictability

The impulse predictability for each MTE shall be $\pm$30% for the minimum impulse burn as defined in paragraph 3.2.1.2.3 and shall decrease linearly to $\pm$5% for impulse burns of 25.0 Kgf-sec and greater.

3.2.1.2.5 Specific Impulse

The steady state minus three sigma specific impulse for a MTE steady state burn at initial thrust level conditions and with the propellant at 20^{o}C shall exceed 225 Kgf-sec/Kgm.

3.2.1.3 <u>Low Thrust Engines</u>

The PS Low Thrust Engines (LTE) shall have the following performance characteristics.

3.2.1.3.1 Thrust Level

Each LTE shall provide an initial thrust of 0.127 $\pm$ 10% Kgf. This thrust level shall be achieved under initial tank pressure conditions and at vacuum using 20°C propellant and pressurant. The nominal thrust level upon completion of the mission shall exceed 0.03 Kgf.

3.2.1.3.2 Total Impulse

The PS shall provide a minimum total impulse of 1215.0 Kgf-sec for accomplishing the spacecraft LTE functions. Any single LTE shall have the capability of providing 750 Kgf-sec of total impulse.

3.2.1.3.3 Duty Cycle

In any single operation at initial tank pressure conditions, each LTE shall be capable of a minimum impulse burn of .002 Kgf-sec ($\approx$ 7 ms pulse) and a maximum impulse burn of 10 Kgf-sec. Each LTE shall be capable of 100,000 on-off cycles. Of these, 20,000 cycles shall be initial starts. The LTE's shall not be duty cycle limited.

3.2.1.3.4 Predictability

The impulse predictability for each LTE shall be as shown in Figure 2.

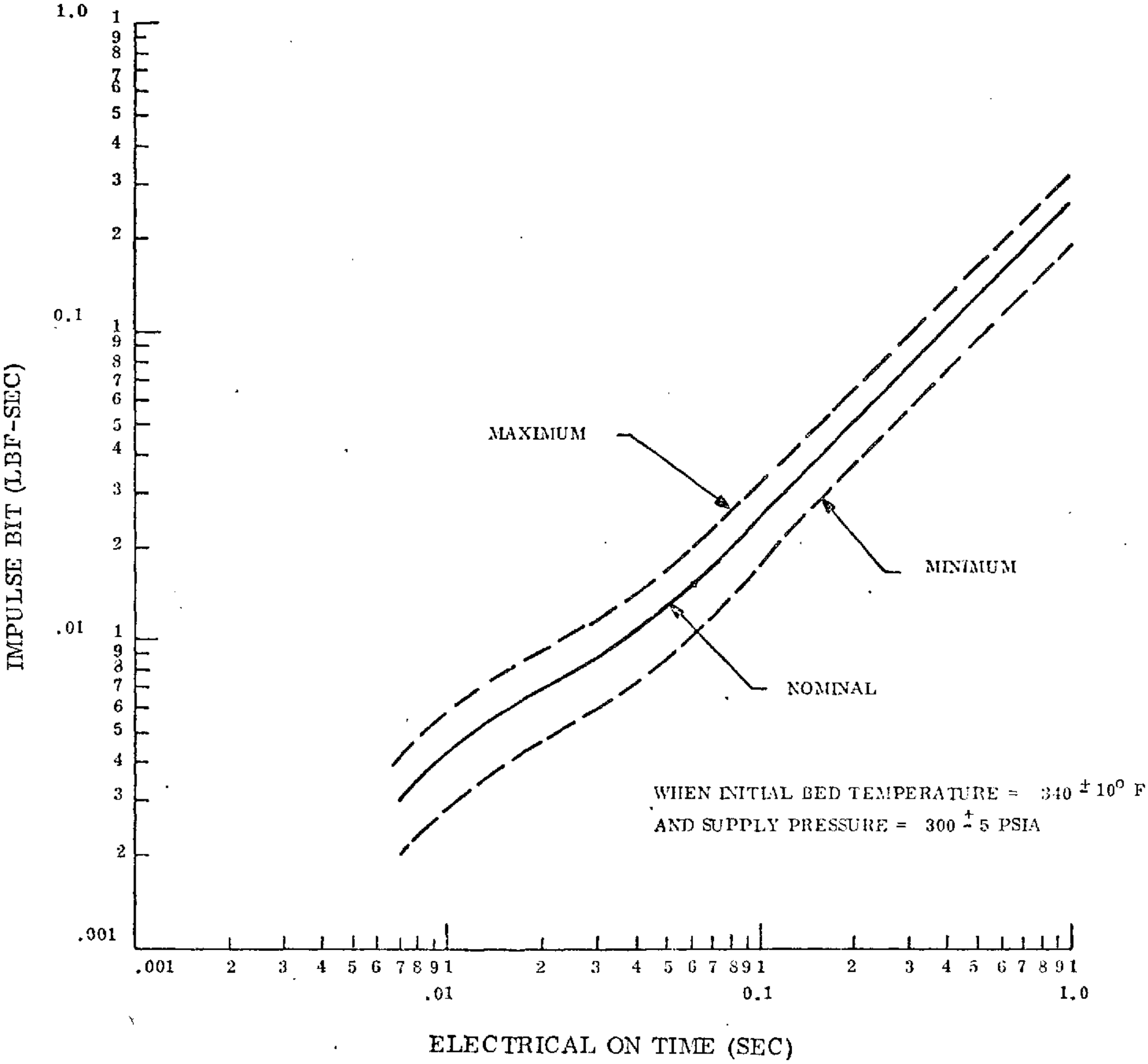
3.2.1.3.5 Specific Impulse

The steady state minus three sigma specific impulse for a LTE steady state burn at initial thrust level conditions and with the propellant at 20°C shall exceed 222 Kgf-sec/Kgm.

3.2.1.4 <u>Engine Alignment</u>

The actual thrust vector of each engine shall subtend an angle of $\leq \pm .15$ degrees, 3σ variation, with the nozzle geometric centerline under all the operating conditions specified in paragraph 3.2.1. The alignment fixture tolerance (i.e.,

Figure 2. Low Thrust Engine Impulse vs On Time



the uncertainty in knowledge of the nozzle geometric centerline angle with respect to the alignment mandrel mirror reference), shall be $\leq \pm.1$ degrees, 3σ variation. An adjustment range allowing ± 2 degrees rotation of the nozzle geometric centerline about two (2) mutually perpendicular directions with an adjustment resolution of $\pm.05$ degrees shall be provided in the design of each engine mount. The nozzle geometric centerline null shall be coincident with the axes as defined in General Electric Drawing (TBD).

3.2.1.5 Stability

Engine chamber pressure oscillations occurring during an engine operation for the period starting with application of the power signal to the propellant control valve plus 0.1 seconds and ending with the removal of the power signals from the propellant control valve shall not exceed $\pm 25\%$ of the nominal steady state chamber pressure.

3.2.2 PHYSICAL CHARACTERISTICS

3.2.2.1 Configuration

The PS shall be designed in a modular arrangement of components. The PS shall be assembled on a structure assembly (General Electric Drawing No. (TBD)). All component interconnect tubing joints shall be welded or brazed.

3.2.2.2 Weight

The dry weight of the PS shall not exceed 37.1 Kg. This weight does not include the PS Structure Assembly. The PS shall have the capability of being serviced with 102 Kg of hydrazine plus the required weight of nitrogen pressurant.

3.2.2.3 Size

The PS size shall be contained within the space envelope shown on General Electric Drawing No. (TBD).

3.2.2.4 Leakage Rate

The total external leakage rate of the PS shall not exceed 30 SCC/hr of gaseous nitrogen when the PS is subjected to system operating pressures.

3.2.2.5 Structural Pressures

The PS shall be capable of withstanding a proof pressure of 1.5 times, and a burst pressure of not less than 4.0 times the maximum operating pressure, except the propellant tank, which shall have a burst to operating pressure ratio of 2 to 1 minimum at maximum operating pressure and maximum temperature.

3.2.2.6 Storage, Transportation, Handling, Assembly and Checkout

The PS shall be capable of performance in accordance with 3.2 herein, after exposure to the applicable environments of paragraph 3.1 of GE Specification (TBD) or the following times:

Transportation	6 weeks
Storage	3 years
Handling and Assembly	3 months (system level)
Test and Checkout	6 months (system level)

3.2.2.7 Design Life

The design life of the PS shall be a minimum of seven years starting with acceptance of the PS by General Electric. The design life of individual components within the PS shall include additional time accrued prior to incorporation into the PS (i.e., transportation, handling, storage, and testing at the component level). The design life of the PS shall consist of an accumulation of the preoperational phase of 3.2.2.6 followed by three years of operational life.

3.2.2.8 Cleanliness

To insure proper performance of the PS, all components, and the subsystem itself, shall meet the cleanliness requirements of Table II. In addition, no metal particles shall be allowed which are over fifty (50) microns.

Table II

Propellant Particulate Cleanliness Requirements

Size Range (Microns)	Maximum Particles Allowed Per 100 Milliliter Sample
5-10	1200
11-25	200
26-50	50
51-100	5
Over 100	0

3.2.2.9 Factors of Safety

The following factors of safety shall be applied to the maximum anticipated applied loads to obtain the respective limit loads and ultimate loads for design/analyses purposes. Limit: 1.15. Ultimate: 1.25.

3.2.2.9.1 Limit Loads

The component shall be designed for sufficient strength to withstand simultaneously the limit loads and other accompanying environmental phenomena given in GE Document (TBD) for each design condition without experiencing yielding or excessive elastic deformation.

3.2.2.9.2 Ultimate Loads

The component shall be designed to withstand simultaneously the ultimate loads and other accompanying environmental phenomena without failure. Failure is defined as structural collapse, rupture, or other inability to sustain the ultimate loads.

3.2.2.10 Wiring and Connectors

All harness wiring and connectors shall be in accordance with General Electric Document (TBD).

3.2.2.11 Dielectric Strength and Insulation Resistance

There shall be no evidence of dielectric breakdown when the PS harness insulation is subjected to 600 volts ac. Insulation resistance shall be a minimum of 50 megaohms when measured at 100 volts dc between all mutually insulated parts and ground.

3.2.3 RELIABILITY

The three-year probability of the PS meeting the performance requirements of this specification shall be (TBD). Reliability apportionments for components of the PS shall be determined by the propulsion subcontractor.

3.2.4 MAINTAINABILITY

The PS shall be designed for ease of maintainability to minimize equipment downtime during assembly, test, and checkout.

3.2.4.1 Maintenance and Repair Cycles

The PS shall be designed such that periodic maintenance will not be required.

3.2.4.2 Service and Access

Access shall be provided to the pressurant and propellant fill and drain valves and the electrical harness interface connectors. Provision shall be made to verify operation of redundant paths without PS disassembly. Provisions shall be made to measure PS external leakage and internal leakage of pneumatic components without PS disassembly.

3.2.5 ENVIRONMENTAL CONDITIONS

The PS shall be designed to withstand the environmental conditions specified in Paragraph 3.1 of GE Document (TBD).

3.2.6 TRANSPORTABILITY

The design of the PS shall be such that the subsystem will meet all performance requirements stated in Section 3 of this specification after the subsystem in its shipping container is subjected to the transportation environments described in GE Document (TBD). The PS shall be designed for shipment by highway (common carrier) and/or aircraft. The PS will not be serviced with propellant during transportation.

3.3 DESIGN AND CONSTRUCTION

3.3.1 MATERIALS, PROCESSES AND PARTS

3.3.1.1 Selection of Electronic Parts

Electronic parts shall be selected in accordance with GE Document (TBD). Parts not on this list shall be submitted to GE for approval.

3.3.1.2 Selection of Materials and Processes

Selection of materials and processes shall be in accordance with GE Document (TBD). Materials and processes not on this list shall be submitted to GE for approval.

3.3.1.3 Standard and Commercial Parts

No commercial parts shall be used without prior General Electric Company approval.

3.3.1.4 Moisture and Fungus Resistance

Where ever possible, non-nutrient materials which resist damage from moisture and fungus shall be used in the PS design. Protective coatings shall not be acceptable as moisture and fungus preventatives for parts which may lose their coating during the normal course of assembly, inspection, maintenance and testing. The requirement of MIL-STD-454C, Requirement 4, shall apply.

3.3.1.5 Corrosion of Metal Parts

The use of dissimilar metals, as specified in MIL-STD-454L, Requirement 16, shall be avoided wherever possible. Materials, techniques, and processes shall be selected and employed with regard to heat treatment procedure, corrosion protection, finish, and assembly and installation such that sustained or residual surface tensile stress, stress concentrations, and the hazards of stress corrosion, cracking, and hydrogen embrittlement are minimized. Processes and materials for protection against corrosion of metal parts shall be selected from those specified in paragraph 3.3.1.2, with the exception that cadmium plating shall not be used. Selected finishes shall be compatible with the thermal requirements of this specification. Materials and surfaces whose compatibility with hydrazine has been established shall be used for parts subjected to long term exposure to hydrazine. The subsystem shall also be internally compatible with deionized or distilled water, isopropyl alcohol as specified in TT-I-735 diluted with 2-6 percent (volume) of water, gaseous nitrogen MIL-P-27401B, air helium MIL-P-27407 Supplement I or Argon MIL-A-18455B.

3.3.1.6 Protective Treatment

All parts shall be corrosion resistant or have a suitable corrosion resistant protective coating applied.

3.3.2 ELECTROMAGNETIC COMPATIBILITY

Electrical and electronic components of the PS shall comply with GE Document (TBD). Compliance with these requirements shall be verified by test or accomplished by proof of similarity.

3.3.3 NAMEPLATES AND PRODUCT MARKING

a. The PS shall be marked for identification in accordance with the manufacturer's standards. The identification shall include, but not be limited to, the following:

1. Nomenclature
2. Customer Part Number
3. Serial Number (Engineering models will use a different designation than prime hardware)

4. Contract Number

5. Manufacturer's Name or Trademark

6. Date of Manufacturer (month, day, year)

b. Hardware or equipment which is not suitable for use in flight, and which would be accidentally substituted for Flight or Flight Spares Hardware shall be red striped with material compatible red paint to prevent such substitution. In the event the hardware is too small to be easily striped, or if test results would be affected by striping, a conspicuous red tag marked "NOT FOR FLIGHT USE" shall be attached.

c. Wire and Cables. Wires and cables for hardware shall not be identified by hot stamping directly onto primary or secondary (shield) insulation.

3.3.4 WORKMANSHIP

The PS including all parts and assemblies, shall be constructed, finished and assembled in accordance with highest standards. Workmanship criteria shall comply with MIL-STD-454C, Requirements 9 and 24. Particular attention shall be paid to neatness and thoroughness of soldering, wiring, marking of parts and assemblies, plating, painting, machine screw assemblage, and freedom from burrs and sharp edges. Electrical soldering shall be per requirements of NHB 5300.4 (3A).

3.3.5 INTERCHANGEABILITY

Each subassembly of the PS and each PS shall be directly interchangeable with regard to form, fit, and function with other subassemblies of the same part number. The requirement of MIL-STD-454C, Requirement 7, shall apply.

3.3.6 SAFETY

The PS shall be designed to limit hazards to personnel and equipment. Explosive and toxic hazards shall be defined and procedures for limiting their effect on personnel and equipment shall be formulated and enforced. The requirements of AFWTRM-127-1 Western Test Range Safety Requirements shall apply.

3.4 MAJOR COMPONENT CHARACTERISTICS

The PS shall consist of the following components of the type indicated. Component requirements shall be further defined to assure that subsystem performance will conform to the requirements of this specification.

3.4.1 PROPELLANT TANK ASSEMBLY

The propellant tank shall be constructed of Titanium 6AL4V, and have a minimum volume of 150,750 cubic cm. Orientation of the propellant at the tank outlet port shall be accomplished by means of either a rubber diaphragm or a surface tension device. It shall be capable of containing hydrazine (per MIL-P-26536), and shall be capable of being pressurized with gaseous nitrogen (per MIL-P-27401) to a nominal operating pressure consistent with the PS pressure schedule. The tank shall be capable of withstanding a reverse ΔP equal to one sea-level atmosphere. Provision for mounting the tank to the PS structure shall be included, together with weldable (or brazeable) fittings for attachment of propellant and pressurant lines. When mounted on the PS structure (oriented as shown on GE drawing (TBD)), the tank shall be capable of supplying propellant to the engines while operating under the orbital acceleration environment specified in Table III. Expulsion of propellant shall be at least 99 percent efficient.

Table III

Lateral Acceleration	(TBD) g's
Longitudinal Acceleration	(TBD) g's

3.4.2 ROCKET ENGINE ASSEMBLY

The engine assemblies shall be actuated by supplying electrical power to a normally closed propellant control valve. Each engine shall contain a combustion chamber, a catalyst bed, an expansion nozzle, a propellant injector, a propellant control valve and provision for mounting. Each engine shall employ hydrazine propellant per MIL-P-26536. A heater shall be incorporated to warm the thrust chamber catalyst bed. The propellant control valve shall provide means for direct attachment to the thruster and contain provisions for welding or brazing to the propellant line. The propellant control valve design shall permit leakage testing of each individual seat. Internal leakage of the valve seat shall not exceed 5 scc/hr GN_2 at operating pressure. The valve shall be capable of continuous power application under conditions of no-flow with no resultant damage.

3.4.3 LATCHING VALVE

The latching valve shall be electrically actuated and shall contain a latching device to maintain itself in the last energized position. The valve shall be used in the propellant lines for isolation of groups of subsystem engines. The valve shall incorporate a switch for position indication, shall utilize welded or brazed inlet and outlet connections, and shall be designed for no less than 1,000 cycles from closed to open to closed.

3.4.4 FILL AND DRAIN VALVE

The fill and drain valve shall be manually operable, and shall be used for filling and draining/venting of the hydrazine or gaseous nitrogen. The valve shall provide non-interchangeable connections for pressurant and propellant usage. The valve shall contain redundant seals for external leakage and utilize welded or brazed tubing connections. The valve shall be capable of 250 opera-

tional cycles from closed to open to closed. Total external leakage of the valve, including the seat, shall not exceed 1×10^{-6} scc/sec helium at operating pressure.

3.4.5 FILTERS

A screen filter shall be supplied in the upstream portion of each thruster and latching valve. These filters shall be compatible with hydrazine and all other fluids and gases to be used in the subsystem. The maximum particle size allowed to pass shall be compatible with the thruster and latching valves.

A system filter shall be provided with filtration to a 10 micron absolute level, and this filter shall utilize welded or brazed tubing connections. Filter size shall be such that trapped dirt equal to 10 times the maximum allowed subsystem contamination will not raise the differential pressure across the filter by more than 3.5 newtons/cm^2. An etched-disc type filter is preferred.

3.4.6 PRESSURE TRANSDUCERS

An absolute pressure transducer shall be supplied in the hydrazine feed line of the subsystem. Pressure measurement tolerances shall be $\pm 1\%$ full scale maximum. Pressure measurement limits shall be a minimum of 0 and 1.25 times maximum operating pressure. The pressure transducers shall not require recalibration when subjected to proof pressure.

3.4.7 PROPELLANT AND PRESSURANT MANIFOLD

Manifolding joints shall be welded or brazed. Stainless steel plumbing is preferred. Manifolds shall be designed and structurally secured to prevent excessive flexing and fatigue during vibration. No braze material shall be used which is catalytic to the propellant or to the precipitation of dissolved salts in the propellant.

3.4.8 THRUST CHAMBER HEATERS

Heaters shall be provided on all engine thrust chambers where necessary to meet the performance requirements of this specification. Dual heater elements shall be provided. Each heater shall be able to heat the thruster to and keep it at the required temperature. Nominal preheat time from $5^{o}C$ to a nondegraded holding temperature shall be a maximum of 100 minutes.

3.4.9 ELECTRICAL INTERFACE PANEL

The electrical interface between the PS and the spacecraft shall consist of a panel which contains four electrical connectors. These provide an interface between the PS and spacecraft wiring harnesses as listed below:

a) Connector 1 - Supplies power to the solenoid valves of all HTE's, MTE's and LTE's

b) Connector 2 - Supplies power to and receives an output signal from all HTE, MTE and LTE temperature sensors

c) Connector 3 - Supplies power to the thrust chamber heaters of all HTE's, MTE's and LTE's

d) Connector 4 - Supplies power to all latching valves; supplies power to and receives an output signal from all latching valve position indicating switches, from the pressure sensor, and from the propellant tank temperature sensor.

3.4.10 WIRING HARNESS

The PS wiring harness shall be designed per (TBD) and shall connect the spacecraft electrically with the various components of the subsystem. Where possible, pigtails shall be employed at the components rather than mateable connectors. The subsystem wiring harness shall, wherever possible, follow the routing of the PS manifolding.

4.0 QUALITY ASSURANCE PROVISIONS

(TBD)

SECTION 8.0

SPECIFICATION

FOR

EARTH OBSERVATORY SATELLITE

SOLAR ARRAY ASSEMBLY

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SECTION 1.0

SCOPE

This specification defines the performance, design and test requirements for the solar array assembly for the EOS spacecraft. One (1) such assembly constitutes the complete solar array for the spacecraft. Each assembly consists of "rigid" panels which are hinged together. Launch retention devices are to be provided to retain these panels in the folded configuration during launch, provide dynamic isolation (if required) to limit vibration induced loads during launch, and provide for the reliable release of the panels. A deployment/retraction mechanism shall erect the solar array assembly, control the rate of motion during erection, lock/rigidize the array when fully deployed, and retract the solar array assembly to a folded configuration upon command.

In the orbital deployed configuration the solar array assembly is driven about the spacecraft roll axis so that the minimum solar angle of incidence is maintained at all times. The solar array orientation drive is not part of the equipment covered by this specification.

SECTION 2.0

APPLICABLE DOCUMENTS

The following documents, of the exact issue shown, form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and the detailed content of this specification, this specification shall be used as the superseding document.

SPECIFICATIONS

GE-SSO

R2761 Rev. E	Wire, Electrical, Insulated, Radiation and Vacuum Resistant, 600V
R4301 AN-1	Connectors, Subminiatures, Space Flight Use for
(TBD)	Cartridge, Electroexplosive Device, Requirements
(TBD)	Electrical Interface Characteristics

STANDARDS

Military

MIL-STD-810B Notice 4 21 September 1970	Environmental Test Methods for Aerospace and Ground Equipment
MIL-STD-454C 15 October 1970	Standard General Requirements for Electronic Equipment
MIL-STD-202D 14 April 1969	Test Methods for Electronic and Electrical Component Parts

NASA

NHB 5300.4(3A)	Requirements for Soldered Electrical Connections

GE-SSO

S30042AB/1	Wire and Lead Preparation, Requirements for
S30042AB/2	Soldering Termination, Preparation and Requirements
S30042AB/4	Harness Assemblies, Requirements for
S30042AB/5	Harness Installation Standard Procedure
S30042AB/6	Termination, Preparation and Crimping Requirements
S30042AB/10	Point to Point Wiring, Requirements for

DRAWINGS

GE-SSO

(TBD)	Solar Array Interface - EOS

OTHER PUBLICATIONS

Optical Coating Laboratory, Inc.

6024000 Rev. A 27 April 1973	Solar Cell Cover, Product Specification

GE-SSO

(TBD)	Approved Materials and Processes List - EOS
(TBD)	Approved Parts List - EOS

Military

MIL-HDBK-5A	Metallic Materials and Elements for Aerospace Vehicle Structures

SECTION 3.0

REQUIREMENTS

3.1 PERFORMANCE REQUIREMENTS

3.1.1 GENERAL REQUIREMENTS

The solar array assembly shall perform the following functions:

a. Convert incident solar energy to electrical energy over the complete operational lifetime.

b. Supply current to the solar array bus at the system operating voltage

c. Provide electrical isolation by the use of diodes so that a short-circuit condition between any part of a solar cell circuit and ground does not load the remaining solar cell circuits

d. Provide telemetry outputs for the purpose of monitoring solar array temperature, deployment actuator temperature, and deployment status

e. Provide protection for the solar cells from the degradation effects of charged particles encountered during the operational lifetime

f. Provide a test connector on each panel for the purpose of testing the telemetry functions and the individual solar cell circuits at the solar cell side of the isolation diodes

3.1.2 ELECTRICAL REQUIREMENTS

3.1.2.1 General

The solar array panel assembly shall utilize 20 mm x 40 mm (nominal size)

solar cells in electrically paralleled solar cell circuits, each isolated from a common positive bus with parallel redundant blocking diodes.A shunt tap point shall be suitably located on each curcuit. These circuits shall be mounted on rigid panels which shall be hinged together to form the solar array assembly. All panels shall be identical and functionally interchangeable.

3.1.2.2 Output Power

At the end of the operational lifetime of the spacecraft, the solar array assembly shall provide 916 watts of electrical power at +29.0 vdc, measured at the array interface connector. This power output shall be available at the autumnal equinox time-of-year with normal solar incidence and at the maximum panel steady-state temperature.

3.1.2.3 Solar Cells

a. Electrical Requirements

The solar cells employed in the manufacture of the solar array panel assembly shall be shallow diffused, N on P silicon solderless cells with a base resistivity between 1 and 3 ohm-centimeters and with palladium passivated titanium-silver N and P contacts. The "N" contact shall be a conventional bar configuration along the long edge of the cell. These cells shall be procured with an individual shipping lot minimum average output current of 250 milliamperes per cell at 0.480 volts, measured at 28°C under equivalent air mass zero, 1 A.U. illumination. Individual cell current at 0.480 volts shall be 237 milliamperes minimum.

The above specified electrical performance requirements shall apply to the cells when covered with the coverglass specified in paragraph 3.1.2.4.

b. Mechanical Requirements

The solar cells shall conform to the mechanical standards of Table 1. Each cell shall fit inside a perfect rectangle of the maximum cell dimensions and cover a perfect rectangle of the minimum cell dimensions excluding corner and edge defects. The "P" contact may have a 1.25 mm maximum void, or "picture frame", along each cell edge at the manufacturer's option. The "N" bar contact may also have a 0.25 mm maximum void along its three edges at the manufacturer's option.

Table 1. Mechanical Standards for Solar Cells

Cell Characteristic	Nominal Measurement	Tolerance
Width	20.0 mm	$\pm$ 0.08 mm
Length	40.0 mm	$\pm$ 0.08 mm
Thickness	0.25 mm	$\pm$ 0.05 mm
Bar Contact Width	0.9 mm	Maximum
Weight	49 gm/100 cells	Maximum

The following solar cell defects shall be considered cause for rejection:

(1) Any crack in the cell

(2) Any twin plane in the cell

(3) Any deep scratch on the cell surface which may be mistaken for a crack

(4) Any combination of edge chips exceeding 5% of the cell perimeter

(5) Any edge chip projecting more than 0.6 mm from the edge of the cell into the face of the cell

(6) Any combination of corner chips on a single cell for which the hypotenuse sum is greater than 2.0 mm

(7) Any gap separation, corrosion, or inclusion in the "N" contact

(8) Any combination of grid gaps exceeding 3% of the total grid length

(9) Any looseness of the grids or contacts which will tend to cause separation

(10) Discoloration or blemishes on the cell surface or in the cell coating which will reduce reliability, make application of the coverglass difficult, or make inspection of the solar array sub-panel assemblies difficult

(11) Any more than 1/3 of the cell grids with gaps in the half of the grid adjacent to the "N" contact

(12) Any bare silicon or titanium area included within the "P" contact

(13) Any inclusion of gas or foreign bodies in the "P" contact, the "N" contact, or the grids. This includes defects sometimes referred to as "blisters" in the contacts.

c. Anti-Reflective Coating

An anti-reflective coating of tantalum pentoxide (Ta_2O_5) shall be applied to the active surface of each cell after contact deposition to reduce incident energy reflections. The "N" contact shall be at least 75% free of anti-reflective coatings. The cell coating shall be uniform and continuous, and fall within color, void and stain limits set up by the manufacturer and approved by GE prior to the start of production.

The anti-reflective coating shall be capable of withstanding the tape-pull test, as defined in paragraph 4.3.1.5, with no apparent degradation.

d. Humidity Resistance

The solar cells shall exhibit no more than a 2% decrease in current at 0.480 volts following exposure to an environment of 90 $\pm$ 5 percent relative humidity at 25 $\pm$ 5°C for a duration of two weeks.

e. Contact Mechanical Integrity

Each solar cell contact shall possess a peel strength of 4.9N minimum as measured using the procedure defined in paragraph 4.3.1.4.

f. Radiation Resistance

The solar cells shall exhibit radiation resistance as defined by the requirements of Figure 1 when subjected to normally incident 1 Mev electron irradiation.

3.1.2.4 Solar Cell Coverglass

Protection against the effects of bombardment by charged particles and solar

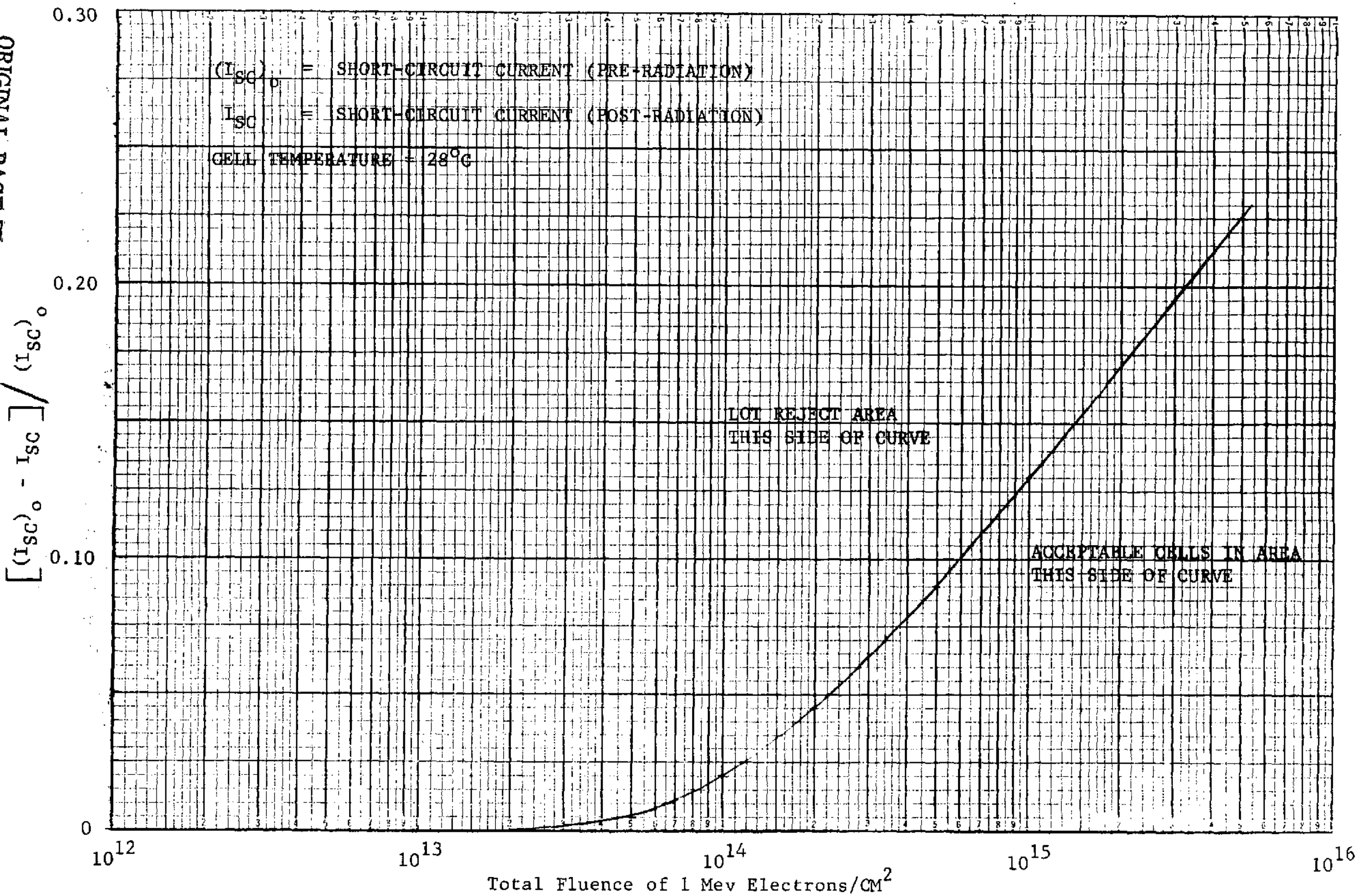

Figure 1. Solar Cell Radiation Resistance

ultraviolet radiation during the array lifetime shall be provided by either covering type 7940 fused silica which meets the requirement of OCLI Specification No. 6024000-03 or by 5% ceria stabilized Chance - Pilkington glass.

The coverglass shall be 150 $\pm$ 15 μm thick. The size shall be specified to ensure minimum exposed silicon area consistent with the requirements of paragraph 3.1.2.6. The cut-on wavelength for ultraviolet rejection shall be 350 nm.

3.1.2.5 Isolation Diodes

Each solar cell circuit shall be diode-isolated from other circuits on the array using two diodes connected in parallel for each circuit. Each diode of a pair shall be capable of passing 1.50 times the maximum solar cell circuit output current continuously.

a. Forward Voltage Characteristics

The forward voltage drop across a diode pair shall not exceed 1.0 volts for a forward current which is 1.50 times the maximum solar cell circuit output current with an ambient temperature of 25 $\pm$ 5°C. The ratio of the current passing through each diode of a diode pair shall not exceed 10 to 1 for an ambient temperature of 25 $\pm$ 5°C and a total forward current which is 1.50 times the maximum solar cell circuit output current.

b. Reverse Current Characteristics

The reverse current through each diode pair shall not exceed 1 milliampere at a reverse voltage of 50 $\pm$ 2 volts for an ambient temperature of 25 $\pm$ 5°C.

3.1.2.6 Coverglass Installation

The coverglass shall be securely bonded to each individual solar cell so as to

insure an environmentally stable lamination. Dow Corning R63-489 adhesive (or approved equivalent) shall be used. Air pockets in the adhesive shall be avoided. Adhesive bubbles of less than 1.0 mm diameter, voids, and delaminations are acceptable provided their total accumulated area does not exceed 40 mm^2 on any cell. Bubbles with diameters less than 125 μm shall be disregarded. The upper surface of the coverglass, and the "P" contact surface shall be free from coverglass adhesive. The coverglass shall be oriented with respect to the solar cell, by means of a Dykem stain, to allow differentiation between the coated surfaces to insure proper installation. The ultraviolet multilayer interference filter shall be bonded to the cell surface such that, if this page were the solar cell with the "N" contact strip at the top of the page, the Dykem stain would be consistently in either the upper right or the lower left corner. The coverglass may protrude over the "N" bar contact provided that at least 0.50 mm is allowed for interconnector soldering. In no case shall there be a gap between the coverglass edge and the cell "N" bar contact strip. There shall be no exposed cell active area along the side opposite the bar contact. The allowable gap along the other two sides shall not exceed 50 μm total. Some overhang of the cell edges is permissible provided it does not exceed 125 μm.

The cells shall be cleaned prior to bonding of the coverglass. Tarnishing or oxidation of the grid lines of the "N" contact is cause for rejection if it extends over more than 5.0 mm. Tarnishing or oxidation of the "P" contact is cause for rejection if it extends over more than 5 percent of the area.

3.1.2.7 <u>Circuit Assembly</u>

a. <u>Output Current Matching</u>

The selection of solar cells which constitute a circuit shall be made based on the output current rating groups. The group output current ratings shall be defined by the solar cell vendor at 0.480 volts in two

milliampere increments. The solar cells shall be appropriately matched to give the required panel electrical performance as defined in paragraph 3.1.2.2. A procedure for assuring this matching shall be prepared and submitted to GE-SSO for approval.

b. Solar Cell Interconnectors

The electrical interconnection of the solar cells shall be accomplished using photoetched silver plated Kovar interconnectors. The interconnector material thickness shall be 75 μm maximum. The supplier shall perform analytical calculations to demonstrate the suitability of the interconnector design for this application. This analysis shall account for the flexure fatigue of the interconnector, and the thermally induced stresses at the solder joints over the temperature range from +75°C to -170°C.

The supplier shall specify the interconnector configuration. The design, along with supporting experimental data and analyses, must be approved by the GE-SSO cognizant engineer.

c. Soldering

Soldering of the interconnector shall be accomplished with minimum solder usage so as to avoid strains on the contacts due to differential thermal expansions between the interconnector material, the solder and the silicon. In order to insure integrity of solder connections throughout fabrication of the solar array panel assemblies and particularly modules where normal soldering procedures are not applicable and where post fabrication inspection is impossible, solar cell soldering standards shall be established by the supplier and submitted to GE-SSO for approval. These standards shall be described in

enough detail so that an inspection criterion can be established and effected throughout fabrication by the various inspectors. These criteria shall include, but not be limited to:

(1) Type of solder used

(2) Quantity of solder

(3) Application of solder

(4) Heat control

(5) Post solder cleaning

(6) Solder joint defects - This last category should include enough information so that any type of solder joint likely to be found on the cell can be classified in terms of acceptance or rejection.

d. Identification

It shall be possible to identify all cells within a circuit. Suitable records shall be maintained so that it is possible to trace every cell back to the production lot.

e. Minimum Clearances

The minimum gap between any cell and/or coverglass and adjacent components (cells, interconnectors, or coverglass) within a circuit shall be at least 200 μm.

3.1.2.8 Panel Assembly

a. Substrate Dielectric

The insulating film between the solar cells and metallic panel substrate shall be nominal 50 μm thick Tedlar film material. This dielectric film shall be laminated to the entire metallic substrate to form a uniform coating over the entire panel substrate.

There shall be no evidence of dielectric breakdown when the substrate dielectric is subjected to 200 volts a.c. Insulation resistance shall be a minimum of 100 megohms when measured at 50 volts d.c. between all circuit terminals and the panel substrate.

b. Circuit Bonding

Each solar cell circuit shall be bonded to the insulated substrate using a flexible adhesive applied in a controlled manner to permit movement of cells relative to one another during thermal cycling. The process for circuit bonding shall demonstrate at least 75 percent coverage of the cell area.

c. Low Energy Proton Protection

The solar cell bar contact area shall be protected against the damaging effects of low energy protons. All protons with energies of less than 1.0 MeV shall be stopped over 95 percent of the bar contact area.

d. Final Inspection Criteria

At the completed panel level, the following visual inspection criteria shall apply for solar cells and coverglass. (See Section 6.0 for definition of terms.)

SOLAR CELLS

(1) An edge nick or chip shall not extend deeper than 0.6 mm into the cell, or more than 2.5 mm along the edge.

(2) The accumulated length of edge nicks and chips shall not exceed 6.0 mm.

(3) A corner chip is acceptable when each side of the chip is greater than 0.6 mm, but the hypotenuse is less than 1.0 mm, or when one side of the chip is less than 0.6 mm and the other side is less than 2.5 mm.

(4) A corner nick shall not be greater than 0.6 mm by 2.5 mm.

(5) Broken or cracked cells are unacceptable.

COVERGLASS

(1) Edge nicks and corner nicks are acceptable

(2) Edge chips and corner chips are acceptable provided that the accumulated area of these chips does not expose more than 0.5% of the active solar cell area of a single cell. On a per cell basis, no more than 50 cells within any circuit shall expose more than 0.3% of the active solar cell area of a single cell.

(3) Cracks which are less than 1.5 mm in length are acceptable provided they are not located over an adhesive void.

(4) Terminated cracks longer than 1.5 mm are permissable up to a maximum of two per cell.

(5) Unterminated cracks (to any edge) are permissible up to 16 per circuit, but no more than one per coverglass.

e. Wire Bonding

Wires shall be bonded to the panel using a silicone rubber compound in a tacking process. The spacing between tacks shall not exceed 50 mm.

f. Use of Low Outgassing Adhesives

All adhesives and potting compound (except for the coverglass adhesive) shall be of the low outgassing variety which exhibits 1% or less weight loss, and 0.1% or less volatile condensable material (VCM) content on exposure to the thermal vacuum environment of 125C and 10^{-6} torr for 24 hours. The selected adhesives shall be identified by their generic and manufacturer's name and number. All applicable test data and specifications shall be supplied to GE-SSO for approval prior to start of panel fabrication.

3.1.2.9 Electrical Output

The output I-V curve of the array, as verified by the summation of individual panel measurements, shall not be less than an analytically derived solar array I-V curve which represents the minimum beginning-of-life performance required to achieve the specified end-of-life power output. This array minimum I-V curve shall be calculated at AMO, 1 AU illumination, normal incidence, 28°C cell temperature, and at the array interface connector.

3.1.2.10 Wiring and Auxiliary Components

a. Wiring and Connectors

All wiring shall be in accordance with the requirements of GE-SSO Specifications S30042AB/1, /2, /4, /5, /6, and /10. Connectors shall conform to GE-SSO Specification R4301. Wire shall conform to GE-SSO Specification R2761. All other materials, such as potting-compound, lacing cord, etc., shall be selected from those specified in the Approved Materials and Processes List - GE Document (TBD) .

b. Temperature Sensor

One platinum resistance type surface temperature sensor shall be provided on each solar array assembly to permit monitoring of in-flight array temperature.

c. Terminal Board

Each panel of the array shall have a terminal board bonded to the rear surface. This terminal board shall serve as the mounting for the isolation diodes. The terminal board shall be conformally encapsulated, including the terminal posts, with a silicone rubber compound which meets the weight loss and VCM content as specified in paragraph 3.1.2.8(f).

d. Test Connector

A test connector shall be provided on each panel to permit the verification of all electrical requirements on an individual circuit basis (at the solar cell side of the isolation diodes) without the need to utilize the flight connector. This test connector shall conform to GE-SSO Specification R4301.

3.1.2.11 Standard Solar Cells

A group of secondary standards will be established by the solar array panel supplier using a solar simulator energy source. Unglassed standards shall be provided for calibration of the light source to test cells. Glassed and mounted cells shall be fabricated for testing submodules, circuits, and array assemblies. Cells used for fabrication of standards shall be selected from the solar cell procurement for this program and shall be representative of the spectral response and efficiency of the production cells. Standard cells shall be recalibrated periodically or whenever visual or electrical damage is evident.

3.1.3 STRUCTURAL REQUIREMENTS

3.1.3.1 General Design Philosophy

The envelope of the stowed solar array assembly is as shown in GE-SSO Drawing No. (TBD) .

The structure shall possess sufficient strength, rigidity and other necessary physical characteristics required to survive the environmental conditions specified herein. It shall survive these conditions in a manner that does not reduce the probability of the successful completion of the mission. Minimum weight structural design shall be achieved wherever practical, reasonable and consistent with the structural design principles and requirements of this specification.

To minimize weight, the structure shall be designed for the critical flight conditions. The non-flight conditions and environments shall influence the structural design only to the extent that no additional flight weight is required. It shall be necessary to devise means for assembly, handling and transportation that do not require an increase in the solar array weight over that required for the flight conditions.

3.1.3.2 Design Loads - Stowed Configuration

The stowed solar array panel assembly shall be designed for the quasi-static limit loads given in Table 2 and loads resulting from acoustic and dynamic tests specified in paragraphs 3.7.3 and 3.7.4. Vibration isolators may be used to limit system responses to levels less critical than those shown in Table 3. Maximum deflections shall be limited to ± 25 mm in any direction when the array assembly is subjected to the quasi-static loads or the qualification vibration test levels shown in Table 7. The minimum lateral frequency requirement of the assembly, on vibration isolators (if utilized) shall be greater than 8 Hz and panel fundamental modes shall be greater than twice the lateral out of plane system frequency.

The loads shown in Table 2 are limit loads and must incorporate appropriate factors of safety. The design yield load will equal the limit load and the design ultimate load will equal 1.25 times the limit load. Structural elements that are alignment critical will have a design yield of 1.15 times the limit load.

Axis	Maximum Load (g's)
Longitudinal	23.0
Lateral	15.0

Table 2. Stowed Solar Array Assembly Structural Design Loads.

3.1.3.3 Strength and Stiffness Requirements

The solar array structure shall be designed to withstand simultaneously the ultimate loads and other accompanying environmental phenomena without failure. Structural deformation within the elastic limit shall not precipitate structural failure during any design condition and environment at loads less than or equal to ultimate loads. Deflections shall be considered excessive if they:

a. Cause unintentional contact between adjacent components

b. Cause any portion of the array to exceed the dynamic envelope

c. Prevent any component, instrument, or device from accomplishing its intended mission objective

3.1.3.4 Solar Array Drive Shaft Load Restrictions

During vibration testing, the loading on the solar array drive shaft shall be limited to ± 1330 N along the shaft axis, and be limited to ± 2220 N in the plane normal to the shaft at the array attachment fitting.

3.1.3.5 Thermal Effects

The effects of temperature during the launch and orbit phases shall be considered in the structural design of the solar array. Thermal effects on the structure including heating rates, temperatures, thermal stresses and deformations, and mechanical and physical property changes shall be based on a critical design heating environment.

3.1.3.6 Material Properties and Allowables

Metallic material strengths and other physical properties shall be selected from MIL-HDBK-5A. If required, further data shall be obtained from reliable test results of recognized laboratories, reports from government agencies or manufacturers' guaranteed data. Strength allowables and other physical properties used shall be appropriate to the loading conditions, design environments and stress states for each structural member. Probability data (values in B column of MIL-

HDBK-5A) and nominal skin gages will be applicable for multi-load path structures. Minimum properties (values of Column A of MIL-HDBK-5A) and minimum skin gages will be applicable for calculating margins of safety for single load path structures.

3.1.4 DEPLOYMENT REQUIREMENTS

3.1.4.1 General Design Philosophy

The solar array assembly launch retention and deployment retraction devices shall retain the stowed panels during launch and subsequently release and deploy these panels upon command. All these devices shall adhere to the envelope restrictions shown in GE-SSO Drawing No. (TBD) . The solar array assembly shall be capable of full deployment without interference between the array elements and between the array and spacecraft. Upon command the solar array assembly shall retract within the envelop it originally occupied during launch.

3.1.4.2 Deployed Natural Frequency

The fully deployed solar array panel assembly shall have a fundamental "free-free" frequency greater than 0.25 Hz in any direction.

3.1.4.3 Deployment Within Gravity Field

The solar array assembly shall be capable of repeated deployment/retractions (not to exceed 20) within a unity gravity field. During deployment testing, the solar array may be supported and/or counterbalanced to simulate the zero G condition. These ground deployments shall not decrease the probability of a successful orbital deployment and retraction.

3.1.4.4 Deployment Release

Deployment release shall be initiated by means of a redundant pyrotechnic device. The release method employed shall generate no loose mechanical debris, and shall release no gaseous contaminants which in any way cause degradation of other vehicle components or systems.

3.1.4.5 Deployment and Retraction Time

The total time of deployment or retraction from release to final latching shall be 20 $\pm$ 10 seconds over the range of temperatures specified in Paragraph 3.7.2.

3.1.4.6 Latching

Latches shall be provided at each axis of rotation to assure a rigid joint. The latches shall provide a method of eliminating all hinge bearing clearances, and when engaged, shall provide a minimum torsinal stiffness on each hinge line of 1350 N-m/rad.

3.1.4.7 Fluid Device Leak Rate Requirement

The leak rate for fluid-filled devices on the solar array assembly shall not exceed 1×10^{-7} standard cm^3/hr. of helium over the range of temperature specified in Paragraph 3.7.2.

3.2 LIFE

3.2.1 STORAGE LIFE

The solar array panel assemblies shall be capable of meeting all the requirements of this specification after three years storage at +23 $\pm$ 5°C and at a relative humidity of 50% or less.

3.2.2 OPERATIONAL LIFE

The solar array panel assembly shall be capable of generating electrical power for two (2) years in a 775 km circular sun-synchronous orbit. At the end of the period, the electrical power output shall be at least that specified in Paragraph 3.1.2.2.

3.3 INTERFACE DEFINITION

3.3.1 MECHANICAL INTERFACES

3.3.1.1 Stowed Envelope

The stowed envelope restrictions and mounting requirements for the solar array assembly are as shown in GE-SSO drawing No. (TBD) . Deflections due to loading environments shall not exceed the envelope by more than $\pm$ 25 mm.

3.3.1.2 Shadowing

There shall be no shadowing of the solar cell circuits by other parts of the spacecraft.

3.3.1.3 Weight

The total weight of each solar array assembly, including associated launch retention and deployment retraction mechanisms, shall not exceed 37.6 kg.

3.3.2 ELECTRICAL INTERFACES

3.3.2.1 Input Power

If electrical energy is required for deployment or retraction activation, the total energy consumed for this purpose shall not exceed 5 watt-min at 28 $\pm$ 2 vdc for a maximum of 30 seconds. No other power will be provided for this purpose.

If electrical energy is required for heaters on the assembly prior to deployment or retraction, the total power consumption for this purpose shall not exceed 5 watts at 28 $\pm$ 2 vdc. for a maximum period of thirty minutes prior to deployment. No other power will be provided for this purpose.

3.3.2.2 Telemetry

The following telemetry signals shall be provided by the solar array assembly:

1. Panel temperature (one per array).
2. All hinge latches locked (one)

3. Deployment retraction actuator temperature (one)

4. Array stowed (one)

The electrical interface shall be per the requirements of GE-SSO Specification No. (TBD) .

3.3.2.3 EED Firing

The interface of the solar array panel assembly EED cartridges with the firing circuitry is as defined in GE-SSO Specification No. (TBD) .

3.4 ELECTROMAGNETIC COMPATIBILITY

The solar cell circuits shall be arranged in a configuration which minimizes magnetic dipole moments. The solar array assembly shall conform to the applicable electromagnetic interference and susceptibility requirements of GE-SSO Specification (TBD) . In particular, any electro explosive device (EED) utilized shall meet the requirements of Paragraph (TBD) of GE-SSO Spec. (TBD) .

3.5 RELIABILITY

A series/parallel configuration at the solar cell circuits shall be selected which optimizes the electrical output during the array lifetime and provides adequate reliability for open-circuit cell failures. The localized heating (or "hot-spot") failure mode shall be considered in specifying the series/parallel circuit configuration. Redundant isolation diodes shall be employed at the positive end of each parallel solar cell circuit.

The solar array assembly shall have a reliability of (TBD) for deployment and (TBD) for retraction after the full operational lifetime.

Whenever the deployment reliability of the assembly will be enhanced through the application of redundancy, as in the case of dual pyrotechnic cable cutters for

deployment initiation, inclusion of this redundancy shall be of prime importance, within the limitation of this specification.

3.6 MAINTAINABILITY

The solar array assembly shall require no maintenance during the specified operational lifetime and storage period. During the panel fabrication and test cycle, cracked or broken cells shall be removed and replaced with cells having equal or higher output current than the replaced cell.

3.7 ENVIRONMENTAL REQUIREMENTS

The solar array panel assembly shall be capable of performing its required functions after exposure to the environments defined below:

3.7.1 RADIATION

The solar array assembly shall be designed to provide the electrical power output specified in Paragraph 3.1.2.2 after exposure to the particle radiation environments given in Tables 3, 4 and 5. The solar array assembly shall be sized to account for the solar cell degradation due to this radiation environment.

Table 3. Geomagnetically Trapped Electron Environment

Electron Energy, E (MEV)	2 Year Electron Fluence (Particles/cm^2 with energy greater than E)
0	5.0×10^{13}
0.25	8.1×10^{12}
0.5	2.6×10^{12}
1.0	6.2×10^{11}
1.5	2.7×10^{11}
2.0	1.3×10^{11}
3.0	2.1×10^{10}
4.0	1.6×10^{9}
5.0	4.1×10^{8}

Table 4. Geomagnetically Trapped Proton Environment

Proton Energy, E (MEV)	2 Year Proton Fluence (Particles/cm^2 with energy greater than E)
2	6.5×10^{10}
3	5.0×10^{10}
5	3.1×10^{10}
10	1.5×10^{10}
30	6.9×10^{9}
50	5.3×10^{9}
100	3.6×10^{9}

Table 5. Solar Flare Proton Environment

Proton Energy, E (MEV)	2 Year Proton Fluence (Particles/cm^2 with energy greater than E)
2	6.2×10^{9}
3	5.8×10^{9}
5	5.4×10^{9}
10	4.5×10^{9}
30	2.1×10^{9}
50	9.8×10^{8}
100	1.5×10^{8}

3.7.2 THERMAL VACUUM CYCLING

The panels shall be subjected to a thermal-vacuum test as summarized below:

a. Chamber pressure less than 10^{-5}torr.

b. The panel shall be cycled between -170°C and +75°C. The cooldown portion of the cycle shall be by natural radiation to a liquid nitrogen shroud. The cooldown shall be terminated when the coldest thermocouple reads - 170°C.

c. The heating portion of the cycle shall also be by natural radiation heat transfer. The hottest thermocouple at the hot end of the cycle

shall read above + 70°C, but shall not exceed + 75°C. This high temperature plateau shall be maintained for 10 minutes before the cooldown is started.

d. The location of the hottest and coldest thermocouples and control of chamber temperatures shall be verified by a "prototype" panel assembly prior to exposure of the flight panels.

e. Prototype or qualification model panels shall be subjected to 50 cycles as described above. Flight model panels shall be subjected to 5 cycles.

3.7.3 ACOUSTIC NOISE

Each panel shall be suspended in a reverberant noise chamber and subjected to the acoustic noise field specified in Table 6. A minimum of six microphones shall be used for control and verification of the acoustic spectrum. The measured overall levels during test shall be within +3 db and -1 db of the specified value. The octave band levels shall be within ± 3 db of those specified except the bands with center frequencies of 4000 and 8000 Hz which shall be within ± 5 db of those specified. Each panel shall be mounted within a picture frame holding fixture for the test. This holding fixture shall simulate the panel edge restraint in the panel launch stowed configuration. The panel/holding fixture shall be mounted within the chamber on a suspension system whose natural frequency is less than 25 Hz.

3.7.4 VIBRATION

The solar array assembly, in the launch stowed configuration, shall be subjected to the sinusoidal and random vibration test schedules specified in Table 7 and 8, respectively. The stowed assembly shall be attached to the vibration exciter via a rigid fixture. The attachment of the item to the fixture shall simulate the actual attachment to the spacecraft structure. Vibration shall be applied

Table 6. Acoustic Noise Test Levels

Octave Band Center Frequency (Hz)	Qualification		Flight Acceptance	
	Sound Pressure Level (db, ref. .0002 microbar)	Duration (seconds)	Sound Pressure Level (db, ref. 0.0002 microbar)	Duration (seconds)
63.0	130	60 for complete exposure	126	30 for complete exposure
125.0	135		131	
250.0	138		134	
500.0	140		136	
1000.0	141		137	
2000.0	138		134	
4000.0	134		130	
8000.0	129		125	
Overall Level	146		142	

Table 7 Sinusoidal Vibration Test+Levels

Qualification

Lateral		Longitudinal**	
Frequency (Hz)	Acceleration, o-p (g's)	Frequency (Hz)	Acceleration, o-p (g's)
5-15	2.3*	5-15	2.3*
15-20	4.0	15-50	3.0
20-30	12.0	50-80	27.0
30-50	2.3	80-100	5.0
50-70	6.5		
70-100	2.3		

Flight Acceptance

Lateral		Longitudinal**	
Frequency (Hz)	Acceleration, o-p (g's)	Frequency (Hz)	Acceleration, o-p (g's)
5-15	1.5*	5-15	1.5*
15-20	2.6	15-50	2.0
20-30	8.0	50-80	18.0
30-50	1.5	80-100	3.3
50-70	4.3		
70-100	1.5		

* Limit displacement to 13 mm D.A.
** Along Vehicle Thrust Axis
2 Octaves/Minutes Sweep Rate

Table 8 Random Vibration Test Levels

Qualification

Frequency (Hz)	PSD (g^2/Hz)	Duration
20-2000	0.09	2 Minutes Per Axis

Flight Acceptance

Frequency (Hz)	PSD (g^2/Hz)	Duration
20-2000	0.04	1 Minute Per Axis

in three orthogonal directions, one direction being parallel to the thrust axis and one direction being normal to the stowed panel surfaces.

3.8 DESIGN AND CONSTRUCTION

3.8.1 MATERIALS, PROCESSES, AND PARTS

3.8.1.1 Selection of Materials, Processes and Parts

Materials and processes shall be selected from GE Document (TBD), Approved Materials and Processes List. Materials and processes not on this list, or used differently than called for in the list, shall be submitted to GE for approval.

3.8.1.2 Selection of Electronic Parts

Electronic parts shall be selected from GE Document (TBD), Approved Parts List. Parts not listed in this document are non-standard and shall be submitted to GE for approval prior to use on the program.

3.8.1.3 Screening of Parts

Parts shall be screened to the requirements of GE Dcoument (TBD), Approved Parts List. Military parts screened to Established Reliability, JANTXV and JANTX shall not require additional screening.

3.8.1.4 Part Specifications

Part specifications will not be required on parts covered by Military or NASA specifications. All other parts will be purchased to control specifications that include the configuration, characteristics, source and screening requirements.

3.8.1.5 Corrosion Prevention

The use of dissimilar metals such as specified in MIL-STD-454, Requirement 16, shall be avoided wherever possible. Materials, techniques, and processes shall be selected and employed with regard to heat treatment procedure, corrosion protection,

finish, and assembly and installation such that sustained or residual surface tensile stress, stress concentrations, and the hazards of stress corrosion, cracking, and hydrogen embrittlement are minimized. Processes and materials for protection against corrosion of metal parts shall be selected from those specified in paragraph 3.8.1.1, with the exception that cadmium plating shall not be used. Selected finishes shall be compatible with the thermal requirements of this specification. All parts shall be corrosion resistant or have a suitable protective coating applied.

3.8.1.6 Moisture and Fungus Resistance

Materials which are not nutrients for fungus and which resist damage from moisture shall be used wherever possible. The requirements of MIL-STD-454, Requirement 4, shall apply. The use of materials which are nutrients for fungus are not prohibited in hermetically sealed assemblies and in other accepted and qualified uses; such as paper capacitors which are treated with a process which will render the resulting exposed surface fungus resistant.

Protective coatings shall not be acceptable as moisture and fungus preventatives for parts which may lose their coatings during normal course of assembly, inspection, maintenance and testing.

3.8.2 NAMEPLATES AND PRODUCT MARKING

a. The solar array shall be marked for identification in accordance with the manufacturer's standards which are consistent with the environmental requirements. The identification shall include, but not be limited to, the following:

   1. Nomenclature
   2. Customer Part Number
   3. Serial Number
   4. Contract Number
   5. Manufacturer's Name or Trademark
   6. Date of Manufacture (month, day, year)

b. Hardware or equipment which is not suitable for use in flight, and which would be accidentally substituted for Flight or Flight Spares Hardware shall be red striped with material compatible red paint to prevent such substitution. In the event the hardware is too small to be easily striped, or if test results would be affected by stripping, a conspicuous red tag market "NOT FOR FLIGHT USE" shall be attached.

c. Wires and cables for hardware shall not be identified by hot stamping directly onto primary or secondary (shield) insulation.

3.8.3 WORKMANSHIP

The component, including all parts and subassemblies, shall be constructed, finished and assembled in accordance with the highest standards for high reliability aerospace equipment. Workmanship criteria shall comply with MIL-STD-454, Requirements 9 and 24. Particular attention shall be paid to neatness and thoroughness of soldering, wiring, marking of parts and assemblies, plating, painting, machine screw assemblage and freedom from burrs and sharp edges. Soldering of electrical wire connections shall be in accordance with NHB5300.4(3A).

3.8.4 CLEANLINESS

Hardware shall be designed, manufactured, assembled, and handled in a manner to insure the highest practical level of cleanliness. Suitable precautions shall be taken to insure freedom from debris within the hardware, and inaccessible areas where debris and foreign material can become lodged, trapped, or hidden shall be avoided. Hardware shall be designed so that malfunctions or inadvertent operating cannot be caused by exposure to conducting or nonconducting debris or foreign material floating in a gravity-free state. Electrical circuitry shall be designed and fabricated to prevent unwanted current paths from being produced by such debris. Ultrasonic vibration shall not be used as a method for cleaning component

assemblies.

3.8.5 INTERCHANGEABILITY

Each subassembly of the component shall be interchangeable with regard to form, fit and function with other subassemblies of the same part number. Likewise, the component itself shall be directly interchangeable with other serial number components. The requirements of MIL-STD-454, Requirement 7, shall apply.

3.8.6 SAFETY PRECAUTIONS

Warnings and precautions relative to personnel and equipment safety shall be specified in component handling, assembly and test instructions.

SECTION 4.0

TESTS

4.1 CLASSIFICATION OF TESTS

The test programs and inspection of solar array panel assemblies and their components shall be classified as follows:

a. Pre-Qualification Tests.

b. Design Qualification Tests.

c. Flight Acceptance Tests.

The supplier will conduct all functional and environmental tests as specified herein. The prototype solar array assembly will be subjected to Design Qualification Tests and the remaining solar array assemblies will be subjected to Flight Acceptance Tests.

4.2 TEST CONDITIONS

4.2.1 ADJUSTMENTS AND REPAIRS

No adjustments, repairs or maintenance shall be permitted during tests unless approved by GE-SSO.

4.2.2 TEST EQUIPMENT

All measurements shall be made with instruments the accuracy of which conforms to acceptable laboratory standards, and which are appropriate for measurement of environmental conditions concerned. These conducted at the supplier's plant shall utilize instruments and test equipment of which the accuracy shall be verified periodically by the supplier to the satisfaction of GE-SSO.

4.2.3 TEST TOLERANCES

During each test of the solar array assembly, the specified test levels shall be achieved within the following tolerances, unless otherwise specified. All measurement equipment shall be calibrated and be sufficiently accurate to provide assurance of meeting the test requirements.

a. Temperature: $\pm$ 2°C ($\pm$ 3.6°F).

b. Vibration Level: $\pm$ 1 db (wideband true rms) for random vibration tests, and $\pm$ 10 percent on sine amplitude.

c. Random Vibration Spectral Shape: Match to spectral shape shall be within + 3 - 1.5 db over the specified frequency range.

d. Time: $\pm$ 5 percent or $\pm$ 10 minutes, whichever is the lesser.

e. Frequency: $\pm$ 5 percent.

f. Relative Humidity: $\pm$ 5 percent.

g. Acoustic Spectral Shape: Match to spectral shape shall be within the tolerance bands as specified in paragraph 3.7.3 for any point within a distance of one foot from the test article.

h. Pressure: 0 to -5 percent from atmospheric to 10 percent of atmospheric. At vacuum conditions, tolerances shall be such that a pressure of 1×10^{-5} torr, or less, is assured.

4.3 PRE-QUALIFICATION TESTS

4.3.1 SOLAR CELLS

4.3.1.1 Electrical Performance

All solar cells shall be individually checked electrically and graded for output as described in paragraphs 3.1.2.2(a) and 3.1.2.6(a).

4.3.1.2 Mechanical Inspection

All solar cells shall be mechanically inspected to determine compliance with respect to dimensions, defects, general appearance and weight. The requirements of paragraph 3.1.2.2(b) shall apply.

4.3.1.3 Humidity - Temperature Test

A random sample of 0.5 percent of each solar cell production lot shall be subjected to a humidity-temperature test. The test conditions shall be 90 $\pm$ 5% relative humidity at 25 $\pm$ 5°C, the the test exposure duration shall be two weeks on the first lot and three days on subsequent lots. The test cells shall be properly identified, traceable to their production lot. Current voltage curves shall be measured under equivalent AMO illumination before and after testing. If any of the test cells exhibits a decrease in current at 0.480 volts of greater than 2 percent, the entire production lot shall be rejected and shall not be utilized in panel fabrication. Similar rejection shall result if any single blister or peel larger than 0.8 mm diameter or more than one blister or peel larger than 0.5 mm diameter is detected.

The time span involved in these humidity-temperature tests dictates that these tests be initiated as soon as as possible after cell delivery. None of these test cells shall be used for panel fabrication.

4.3.1.4 Contact Pull Test

An additional 0.5 percent of each cell production lot shall be randomly selected and subjected to a contact pull test. This test shall be performed on both the N and P contacts. The pull test ribbon shall be of the configuration shown in Figure 2. The ribbon material shall be the selected solar cell interconnector material. The ribbons shall be soldered within the areas shown on Figure 3. The ribbon soldering technique shall be determined by the subcontractor as long as it accomplishes the test objectives. The solder joint shall have a minimum

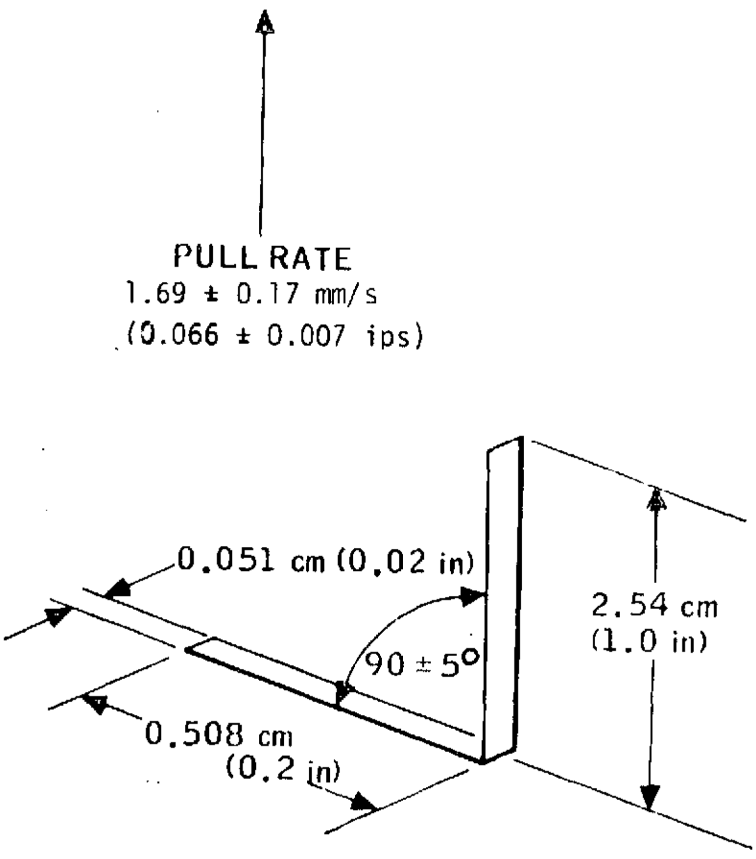

Figure 2. Contact Pull Test Ribbon Configuration

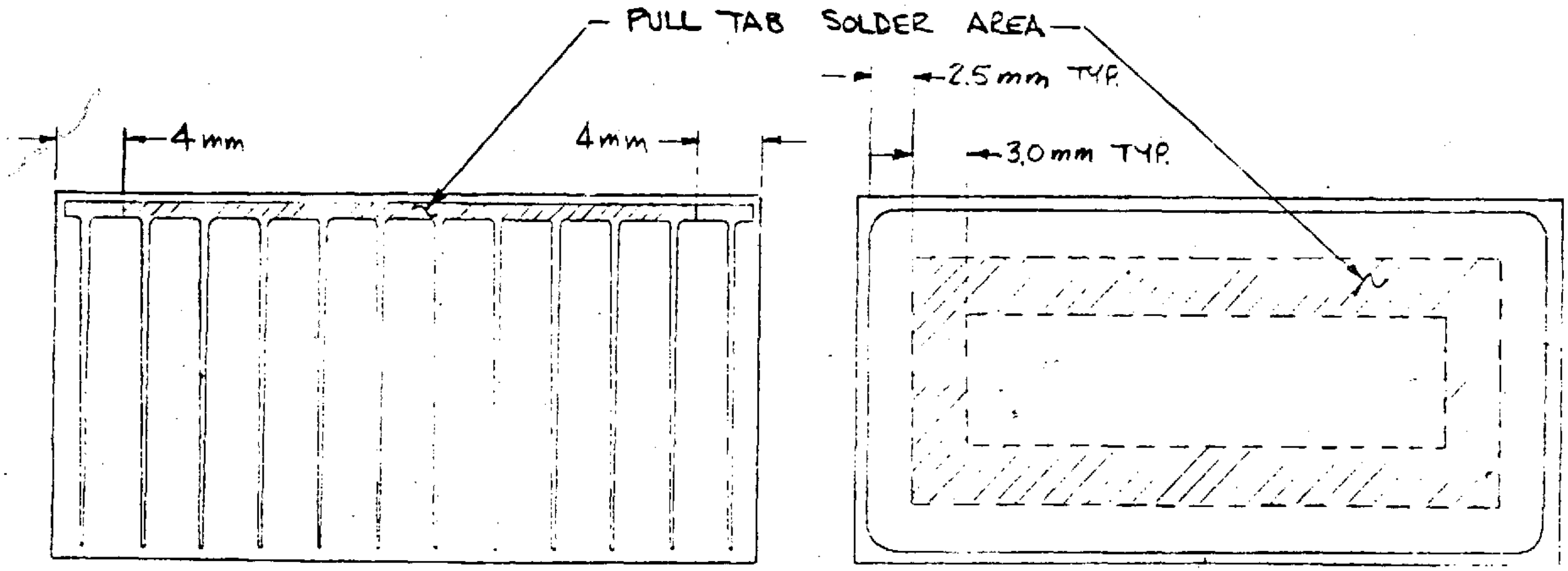

Figure 3. Definition of Area for Pull Test Load Soldering

solder fillet coverage of 95 percent of the available contact area. Defective solder joints such as a cold-soldered joint shall be rejected prior to testing. There shall be no more than two (2) ribbon attachments per contact face.

The cell shall be clamped to prevent movement in such a manner that the restraining force is distributed over at least 40 percent of the cell surface. The tensile force applied to the tab normal to the face of the cell shall be applied at a uniform rate of 1.69 ± 0.17 mm per second. The reading of the tensile force indicator shall be recorded at the instant of separation of the tab from the cells. If any contact separation force is less than 4.9N, another one percent random sample shall be selected from the same production lot and tested as above. Failure of any one of the contact tests of these cells shall be cause for rejection of the production lot. Pulling ofthe tab from the solder shall not be considered a failure provided no contact removal is evident.

4.3.1.5 Tape Pull Test

All cells shall be subjected to a tape pull test on the "N" surface. Scotch No. 810 Magic Transparent Tape shall be used. The tape shall be applied parallel to the "N" contact and pressed to the cell with sufficient force to insure full transparency. The tape shall be stripped from the cell at a rate of 50 to 125 mm per second. For acceptance, the cell shall exhibit no damage to the contact, grid line, or anti-reflective coating. All solar cells must be cleaned after performance of this test to insure that no residue of the tape adhesive remains on the solar cell surface.

4.3.2 COVERGLASS

All coverglass shall meet the acceptance test requirements specified in OCLI Specification No. 6024000.

4.3.3 ADHESIVES

a. In order to provide assurance against defective batches of adhesive, a test sample of each mixture shall be prepared and examined for appropriate physical characteristics.

b. Manufacturer's recommended adhesive shelf life and storage conditions shall be maintained.

4.3.4 COVERED SOLAR CELLS

Following coverglass bonding, each cell shall be visually inspected for mechanical defects and for proper coverglass placement. The requirements of paragraphs 3.1.2.3 and 3.1.2.6 shall apply.

Ten bonded coverglass/solar cell samples shall be submitted to GE-SSO for further evaluation.

4.3.5 CIRCUITS

Each circuit shall be visually inspected prior to bonding to the substrate. All solder joints shall be inspected under 10x magnification to determine defects. The requirements established under paragraph 3.1.2.7(c) shall apply.

4.3.6 ISOLATION DIODES

The solar cell circuit isolation diodes shall be screened in accordance with the requirements of GE Document (TBD). In addition, each diode pair shall be tested to verify the requirements of paragraphs 3.1.2.5(a), and 3.1.2.5(b).

Continuity through each diode shall be verified as part of each panel electrical performance test.

4.3.7 SUBSTRATE DIELECTRIC

The substrate dielectric shall be tested for dielectric strength to satisfy the requirements of paragraph 3.1.2.8 (a). This test shall be performed in accordance with MIL-STD-202, Method 301 using the following parameters:

1. The test voltage shall be 200 volts a.c.
2. The test voltage shall be applied across the wet probe and the panel substrate.
3. Application of the test voltage shall be for 60 seconds with a total test time adequate to cover the entire substrate area.

4.3.8 PANEL ASSEMBLY

4.3.8.1 Insulation Resistance

The component shall be tested for insulation resistance to satisfy the requirements of paragraph 3.1.2.8 (a). This test shall be performed in accordance with MIL-STD-202, Method 202, using the following parameters:

1. The test voltage shall be 50 volts dc.
2. The test voltage shall be applied across each circuit terminal and the panel substrate.
3. Electrification time shall be 2 minutes.

4.3.8.2 Circuit Continuity

Prior to the first illumination test, a continuity test shall be performed on all circuits to insure that all point-to-point wiring is in accordance with the panel electrical interconnection diagram.

4.3.8.3 Visual Inspection

A detailed visual inspection shall be performed on completed solar array panel assemblies. The requirements of paragraph 3.1.2.8(d) shall apply. The data for each inspection of the completed panel assembly shall be submitted to GE-SSO for permanent retention at the time of delivery of the panel.

4.3.8.4 Illumination Test

Whenever a solar cell circuit illumination test is specified, the following test conditions for temperature, illumination intensity and standard cell selection shall

be adhered to if an artificial illumination source is used:

a. Temperature. Each solar array panel shall meet the requirements of paragraph 4.3.8.4(d) when the solar cell circuits are tested at a temperature between 22 and 33°C. The temperature distribution over the circuit area under test shall be uniform within $\pm$ 2°C. If the test temperature is different from 28°C, the resulting I-V characteristic shall be adjusted to account for this temperature difference. The method of adjustment shall be approved by the GE-SSO cognizant engineer.

b. Illumination Intensity. The intensity and spectrum of the light source shall produce a short circuit current in a glassed and mounted secondary standard solar cell which is no greater than 104% nor less than 96% of its calibration value at a minimum of 10 points uniformly distributed over the circuit area under test. The average value of the standard cell short circuit current for the 10 points shall be no greater than 101% nor less than 99% of its calibration value. The standard cell to be used for monitoring intensity shall be selected as per paragraph 4.3.8.4(c).

c. Standard Cell Selection for Illumination Testing. A total of four glassed and mounted secondary standard cells shall be used. Adjust the intensity of the light source on the first cell until measured short-circuit current at the test temperature is the calibrated value. Measure the three other standard cells, one at a time, in the same position as the first cell, under the illuminator without further intensity adjustment. Add the four values of short-circuit current thus measured and compare with the sum of the calibrated short-circuit current values for the four standard cells. The sum of the measured short-circuit current of the cells shall be within 1% of the sum of the calibrated values or the process shall be repeated using another cell to adjust the

illuminator intensity. If none of the four cells will provide an intensity that will meet the above requirement, select the cell that provided a summation of measured values less than, but closest to, the calibrated values for control of illumination intensity. This cell shall be used to adjust the illumination intensity for verification of the requirements of paragraph 4.3.8.4(d). The illumination intensity of the standard cell shall be adjusted by changing the distance between the light source and the standard cell. The voltage of the light source and the illumination intensity on a given solar cell circuit shall not be changed after the standard cell has been selected as described above.

d. Illumination Test Acceptance Criteria. Output characteristics (I-V Curves) of each panel, measured at the array interface connectors, shall be obtained according to the illumination test requirements specified above. Values of output current at specified voltages shall be the electrical acceptance criteria. I-V curves obtained during all illumination tests shall be smooth and continuous, exhibiting a characteristic shape typical of an individual solar cell.

4.3.9 THERMAL CYCLING PANEL SEGMENT

A thermal cycling qualification panel segment shall be fabricated and tested to demonstrate the ability of the interconnect system to survive the anticipated thermal cycling environment. This panel segment shall be representative of the construction to be used on the flight configuration panel. At least 100 cells shall be mounted on this panel. The thermal cycling test shall be preceded by a detailed microscopic (10x) examination of the test panel and an illumination test.

The thermal cycling test shall include a total of 2000 cycles. Each cycle shall be composed of a low temperature limit at -170°C and a high temperature limit at +75°C. The specimen temperature shall be measured by thermocouples embedded between the substrate and the solar cells. The rate of temperature change between limits shall not exceed 100°C/minute.

The qualification panel segment shall be removed from test at 5, 100, 250, 500, 1000, and 2000 cycles for detailed microscopic (10x) examination and an illumination test. Any evidence of interconnect joint separation, silicon flake-off at the interconnect joint or interconnect strand fracture shall be cause for rejection. Any decrease in current at maximum power voltage which is more than two percent of the initial value shall be cause for rejection.

4.3.10 LAUNCH RETENTION AND DEPLOYMENT MECHANISMS

The following tests shall be performed on engineering development components, as specified by generic category below:

a. Damper/Snubbers

   1. Damping force versus temperature over the predicted operating temperature range.

   2. Ultimate strength at minimum, nominal and peak predicted operating temperature.

b. Latch/Release Cables, Spring and Guides

   1. Spring force and rate.

   2. Ultimate strength/preload ratio.

   3. High speed photographic analysis of release sequence.

c. Pyrotechnic Cable Cutter. Single and dual fire release capability in conjunction with release cable test.

d. Deployment Actuators. Deployment torque as a function of temperature and angle (also voltage, pulse rate, etc., if applicable).

e. Deployment Rate Controllers. Rate control torque as a function of temperature, load and rate (also voltage, pulse rate, etc., if applicable).

f. Panel Hinges and Yoke Hinge. Function torque, free motion, stiffness, sensitivity to misalignment at minimum, nominal, and maximum temperatures.

g. Deployment Latches. Reliability of latching with various rates of closure and spring-back torques.

4.3.11 ELECTRO-EXPLOSIVE DEVICE CARTRIDGES

All electro-explosive device cartridges shall be subjected to the test program specified in GE-SSO Specification No. (TBD).

4.3.12 THERMAL VACUUM DEPLOYMENT MODEL

A thermal vacuum deployment/retraction test shall be performed on the first solar array assembly structure which is fabricated. The panels on the model may have mass simulated solar cells, but the structure and launch retention and deployment mechanisms must be of flight configuration. The array assembly shall be secured in the launch stowed configuration. Under vacuum conditions, input heat fluxes shall be imposed to simulate the calculated temperature distribution through the assembly at the time of orbital panel deployment. High speed photographic coverage of the panel release events shall be provided.

4.3.13 STRUCTURAL DYNAMICS MODEL (SDM)

The solar array assembly for the structural dynamics model shall provide 100% solar cell coverage on the exterior panel. These cells are not required to meet the electrical performance requirements of paragraph 3.1.2.3(a). The other panels shall have mass simulated solar cells. The solar cell covered panel shall be subjected to the tests of paragraph 4.4.2. The solar array assembly shall be subjected to the tests of paragraph 4.4.3 in the stowed condition.

The solar array assembly shall be tested in a deployed configuration to verify the requirements of paragraph 3.1.4.2, and to obtain data for mathematical model verification. The deployed solar array assembly shall be cantilevered from a rigid fixture vertically such that the solar array is loaded in plane by +1g. Lateral bending and torsional testing shall be performed by utilizing lateral displacement, quickly released to measure natural frequencies, mode shapes and damping characteristics of the system.

4.4 DESIGN QUALIFICATION TESTS FOR PROTOTYPE PANELS

Each prototype solar array panel shall be subjected to the following sequence of tests:

a. Inspection and illumination.

b. Weight and center-of-mass.

c. Acoustic noise per paragraph 3.7.3

d. Inspection and illumination.

e. Thermal vacuum per paragraph 3.7.2.

f. Inspection and illumination.

4.5 DESIGN QUALIFICATION TESTS FOR PROTOTYPE SOLAR ARRAY ASSEMBLY

The prototype solar array assembly shall be subjected to the following sequence of tests:

a. Inspection and illumination.

b. Weight and center-of-mass.

c. Vibration per paragraph 3.7.4

d. Ambient deployment.

e. Inspection and illumination.

4.6 FLIGHT ACCEPTANCE TESTS FOR FLIGHT PANELS

Each flight solar array panel shall be subjected to the following sequence of tests:

a. Inspection and illumination.

b. Weight and center-of-mass.

c. Acoustic noise per paragraph 3.7.3.

d. Inspection and illumination.

e. Thermal-vacuum per paragraph 3.7.2.

f. Inspection and Illumination.

4.7 FLIGHT ACCEPTANCE TESTS FOR FLIGHT SOLAR ARRAY ASSEMBLIES

Each flight solar array assembly shall be subjected to the following sequence of tests:

a. Inspection and illumination.

b. Weight and center-of-mass.

c. Vibration per paragraph 3.7.4

d. Ambient deployment.

e. Inspection and illumination.

SECTION 5.0

PREPARATION FOR DELIVERY

5.1 PACKING

5.1.1 The supplier shall furnish an individual, reusable shipping container for handling, storage and shipment of each solar array assembly.

5.1.2 The shipping containers shall be capable of meeting the requirements of MIL-STD-810, Methods 514 and 516 (Vibration, Shock and Drop Test).

5.1.3 The assembly, in the launch stowed configuration, shall be secured to a suitable holding fixture which shall then interface with the shipping container. No other portion of the shipping container shall be in contact with the panel.

5.1.4 Each solar array assembly shall be encased in a flexible, water vapor barrier bag before being packaged in its shipping container. (This material shall be suitable and shall not degrade the final solar panel operation in any way.) A relative humidity card and constrained desiccant bags shall be enclosed within this bag for shipment.

5.1.5 Each shipping container shall be marked with the part number and serial number of the solar array assembly it contains. Each side of the container shall be marked "FRAGILE - SOLAR ARRAY" in bold three inch high lettering.

5.2 HANDLING AND TRANSPORTATION

All handling and transportation shall be accomplished with extreme care consistent with the fragile nature of the product. Air shipment is preferred if shipping distance is greater than 400 km.

SECTION 6.0

NOTES

6.1 DEFINITIONS

The definitions listed herein are to be used in establishing uniform nomenclature for use in this specification.

a. Pre-qualification Tests. Tests which are intended to verify the design adequacy of the product prior to ro during the fabrication cycle.

b. Design Qualification Tests. Tests which are intended to verify the design adequacy and to demonstrate a minimum level of acceptable quality. These tests are designed to evaluate performance under a simulated environment which is sufficiently more severe than the actual environment to assure location of faults, yet not so severe to create unrealistic modes of failure.

c. Flight Acceptance Tests. Tests intended to disclose workmanship defects in sufficient time to permit correction prior to the use of the article. The test levels and durations are representative of the environment in order to detect and eliminate early-life failure, yet mild enough to avoid fatiguing or wearing out the equipment.

d. Submodule. A set of solar cells connected electrically in parallel.

e. Circuit. A group of series connected modules which supply current at the solar array operating voltage. Each circuit is diode isolated from other circuits on the solar array panel.

f. Lot. A lot is defined as an accumulation, for shipment, of solar cells manufactured under the same conditions and accepted after performance of the individual acceptance tests. The lot container shall be marked to indicate the production lot numbers included.

g. Limit Loads. The maximum calculated loads that may reasonably be expected to occur for the specified conditions of operation.

h. Ultimate Loads. The product of the limit loads and an ultimate factor of safety. Ultimate loads are used for design, stress analysis and static tests as a means of assuring high probability of functional and structural adequacy under the application of limit loads.

i. Allowable Ultimate Stress. The stress at which catastrophic failure of the structure occurs. A catastrophic failure is defined as a failure producing general collapse or instability of the structure.

j. Allowable Yield Stress. That stress at which the material exhibits a permanent deformation of 0.002 inch per inch (0.2 percent).

k. Factor of Safety. The factor of safety is an arbitrary factor intended to account for minor variations from item to item in material properties, fabrication quality and details, internal load distributions within the structure, and possible degradations in strength that may result from the actual history of treatment of each item in service. It is used to obtain corresponding proof and ultimate loads for use in design, analysis and tests.

l. Margin of Safety. The percentage by which the allowable load (or stress) exceeds the design load (or stress). Margin of safety is defined as:

$$M.S. = \frac{1}{R} - 1 ,$$

where R is the ratio of applied load (or stress) to the allowable load (or stress). Margins of safety shall be computed at both yield and ultimate load levels. All structures shall have a positive margin of safety (i.e., M.S. $\geq$ zero).

m. Chip

The local absence of silicon (or glass) along the edges and corners of a solar cell (or coverglass) which extend through the thickness of the cell (or coverglass).

n. Nick

The local absence of silicon (or glass) along the edges and corners of a solar cell (or coverglass) which do not extend through the thickness of the cell (or coverglass).

o. Terminated Crack

A crack which has only one end extending to an edge.

p. Unterminated Crack

A crack which has both ends extending to an edge.

SECTION 9.0

9.0 SPECIFICATION FOR THE EOS ELECTRICAL INTEGRATION SUBSYSTEM

(This specification is presented in Volume 3, "Specification for the EOS General Purpose Spacecraft Segment and Modules" of Report No. 5, "System Design and Specifications".

Specification No. SVS-XXXX
16 September 1974

SECTION 10

SPECIFICATION

FOR THE

SCANNING SPECTRAL RADIOMETER

(THEMATIC MAPPER)

TABLE OF CONTENTS

SECTION 1.0

THEMATIC MAPPER

1. Scope

This specification is written as a preliminary draft to consolidate EOS Mission & System requirements and guidelines that should be incorporated into the final instrument specification. This includes performance parameters, preliminary interfaces with the spacecraft, and operations. This specification does not include the programmatic, schedule and test requirements that must be included in the instrument contracts.

SECTION 2.0

REQUIREMENTS

2.1 Functional

2.1.1 General

The Thematic Mapper (TM) gathers data by imaging, filtering and detecting reflected solar radiation from the surface of the earth in several spectral bands simultaneously through the same optical system. Spectral definition shall be obtained by spatially separating the available energy and placing multilayer interference filters in the optical path to the detector arrays in the focal plane. Detectors for this radiometer shall utilize state-of-the-art technology.

The TM shall be designed to operate in a three axis stabilized Earth Observation Spacecraft and provide a continuous 185 kilometer wide strip map at nadir in each specified spectral band when in the following orbit:

> The Observatory will operate in a circular, sun synchronous, near-polar orbit at an altitude of 775 Km (418 nautical miles). The local solar time at the north to south equatorial crossing is 1100 $^{+30}_{-0}$ hours. The observatory will complete 14 6/17 orbits per day and views the entire earth every 17 days. The orbit has been selected and will be maintained so that the satellite ground trace repeats its earth coverage at the same local time every 17 day period with a ground swath (perpendicular to the subsatellite track) 185.3 Km wide. A nominal 13% sidelap between adjacent image frames will be provided at the equator.

2.1.1.1 Sensor Sensitivity

The sensor sensitivity shall be adequate to produce high quality signals in each spectral band from low reflectance scenes located in regions between 50^{o} north and 50^{o} south latitude when the sun is 30^{o} above the horizon.

2.1.1.2 Offset Pointing

The option to incorporate offset pointing (up to $\pm 26^{o}$ from NADIR) as a low impact design modification at a later time shall be a design objective of the instrument. This feature will not be implemented on EOS-A but may be incorporated later in the EOS program as a precursor to a sample data system.

2.1.1.3 Operations

An average instrument operation time of 15 minutes per orbit, with a maximum on-time of approximately 30 minutes shall be required to permit the imageing of land masses and near coastal waters on a global basis. A design goal is to eliminate multiple instrument gains and modes so as to reduce operational complexity. Instrument radiometric calibration updates (both relative and absolute) in orbit should be minimized via instrument design.

2.1.1.4 Scan Technique

The scanning technique for the TM may be either image plane or object plane provided the selected approach satisfies the performance requirements set forth in this specification. The cross scan pattern shall be linear. As a design goal, the scan efficiency shall be a minimum of 80%.

2.1.2 Spectral Definition

The total spectral response shall be specified for each band and shall include the optics and the detectors. This response shall be obtained by measurement of both the optical spectral bands and the spectral response of the detectors within and outside the band in which they are going to be used.

2.1.2.1 Filter Characteristics

The spectral filter used to define the six bands shall have the characteristics shown in Table 1.

2.1.3 Instantaneous Fields of View (IFOV)

The elemental instantaneous field of view shall be defined as the solid angle bounded by the points where the detector voltage response to a point source is 50% of the maximum obtained when the source is located on the axis defined by the telescope assembly. The voltage response to this same point source shall be 1% or less at angular distances equal to or greater than two (2) elemental fields of view.

TABLE 1

FILTER CHARACTERISTICS

Band Number	1	2	3	4	5	6
Filter Type	Bandpass	Bandpass	Bandpass	Bandpass	Bandpass	Bandpass
Cut-On Wavelength (μM)	.45 ± 0.01	.52 ± .01	.63 ± .01	0.8 ± .01	1.55 ± .01	10.4 ± .01
Cut-Off Wavelength (μM)	.52 ± 0.01	.60 ± .01	.69 ± .01	0.95 ±.01	1.75 ± .01	12.6 ± .01
Edge Slope (1) (μM)						
Short Wave Side	.02	.02	.02	.02	.02	.02
Long Wave Side	.04	.045	.05	.05	.04	.04
Efficiency (Minimum)	.9	.9	.9	.9	.9	.9
Spectral Flatness (2) (percent)	75	75	75	75	75	75
Spurious Transmittance (3) (percent)	5	5	5	5	5	5

NOTES:

(1) Edge slope is defined as the wavelength interval between 5% absolute transmittance and 70% of peak tranmittance. The above are maximum values.

(2) Spectral flatness is defined as the percent of the wavelength range (bandwidth) over which the transmittance does not vary by more than 5% of the peak transmittance.

(3) Spurious transmittance is the ratio of the integrated solar energy transmitted outside the bandpass to that within the bandpass. This requirement may be waived where it can be shown that the detector response outside the bandpass is less than 5% of the total integrated energy within the bandpass.

Measurement of all elemental fields of view shall be made with the scanning system drive disabled and the scan mechanism locking in a fixed position relative to the housing reference surfaces. The elemental field of view shall be some geometric shape defined by the appropriate number of microradians on a side, if a square, or some number of microradians in diameter if a circle. These measurements shall be accurate to within 4 microradians. The basic requirement of the TM is that it shall have a resolution at the subsatellite point of no worse than 40 meters, with the optics sized for a goal of 30 meters, for the first five bands, from an altitude of 775 kilometers and a cross-track swath width of 185.3 kilometers. The overall objective of the EOS-A System is to provide output product resolution of 40 meters or better. For the seventh band, the above requirements are reduced to 160 and 120 meters respectively. The field stops for bands 1 through 4 may use square optical fibers arranged as compactly as possible about the center of the image plane. Bands 5 and 6 shall use the surface of the detectors as their field stops. The fiber optic spacing in the focal plane of the first four bands shall be such as to allow precisely coincident samples from band to band.

2.1.4 Radiometry

2.1.4.1 Radiation Detection and Signal Amplification

The radiation from the scene shall be focused by an optical system onto detector arrays. The signals from the detectors shall be amplified to the levels required by the digital signal processor. The amplifier shall have a linear output voltage versus input radiant energy transfer characteristic.

2.1.4.2 Minimum and Maximum Radiance and S/N

Table 2 indicates the radiance range to be accommodated by the instrument and the S/N required at minimum radiance.

Table 2. 6 Band TM Performance Matrix

Band	Spectral Range (Microns)	Nominal Ground Resolution (Meters)	Min. Radiance* (mw/cm^2-ster)	Max. Radiance** (mw/cm^2-ster)	S/N at Min. Rad. ***
1	.45 - .52	40	.22	2.24	TBD
2	.52 - .60		.20	2.34	
3	.63 - .69		.09	1.30	
4	.8 - .95		.12	1.79	
5	1.55 - 1.75		.12	1.00	
6	10.5 - 12.6	160	.77 ($243^\circ K$)	2.62 ($320^\circ K$)	NE $\Delta T = .5^\circ K$ at $300^\circ K$

* based on bare soil at $47^\circ N$ in December with ground visibility of 10Km

** scaled from ERTS-1 values

*** required additional user definition

2.1.4.3 Radiometric Error Budget

The instrument allocation of the total system error budget is divided into two segments: fixed and temporal error budgets as shown in Figure 1 and Table 3.

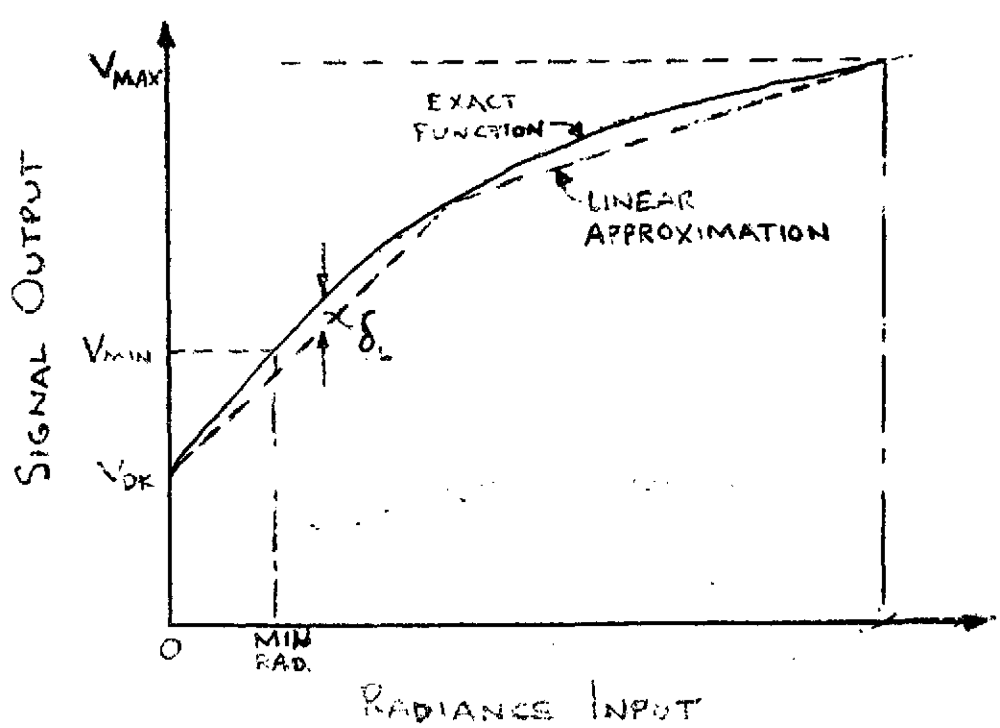

Figure 1. Radiometric Transfer Characteristic

TABLE 3

THEMATIC MAPPER RADIOMETRIC ERROR BUDGET

CHARACTERISTIC	DEFINITION	VALUE	COMMENTS
FIXED ERRORS			
1. Transfer Characteristic Linearity $\left(\frac{\delta L}{V}\right)$	Deviation of approximate Linear Characteristic from exact transfer characteristics over a given linear segment.	5% at min. rad. 0.3% at max. rad.	Required number of linear segments used in the approximation is selected based upon this requirement.
2. Spectral	Band-to-Band inaccuracy in knowledge of spectral band definition and calibration source spectral radiance.	1.6% at any radiance input	Cal. source uniformity shall be optimized to minimize error due to shift of irradiance at focal plane.
3. Spatial	Across field-of-view error in knowledge of irradiance at instrument focal plane as a function of field angle.	0.8% at any radiance input	
TEMPORAL ERRORS			
1. Calibration Source Stability	Long term stability in on-board calibration source spectral radiance.	8% at min. rad. 0.5% at max. rad.	
2. Transfer Characteristic Gain or Responsivity $\left(\frac{\delta R}{R}\right)$	Variation in sensor gain or responsivity between calibration updates for each linear segment.	0.3% gain variation	
3. Transfer Characteristic Offset $\left(\frac{\delta DK}{V_{DK}}\right)$	Variation in dark current or voltage between calibration updates.	$\frac{\delta DK}{V_{DK}} = .05\left(\frac{V_{min}}{V_{DK}} - 1\right)$	

NOTES: Above limits on transfer characteristic errors can be adjusted; however, errors should not exceed the following:

MAXIMUM RADIANCE $$\sqrt{\left(\frac{\delta L}{V}\right)^2 + \left(\frac{\delta R}{R}\right)^2 + \left(\frac{\delta DK/V_{DK}}{(V_{max}/V_{DK})-1}\right)^2} \leq 0.005$$

MINIMUM RADIANCE $$\sqrt{\left(\frac{\delta L}{V}\right)^2 + \left(\frac{\delta R}{R}\right)^2 + \left(\frac{\delta DK/V_{DK}}{(V_{min}/V_{DK})-1}\right)^2} \leq 0.08$$

2.1.4.4 Radiometric Calibration

Radiometric Calibration is intended to establish and monitor the absolute radiometric sensitivity of the instrument to meet the requirements of the preceeding paragraph. To this end, the following design and operations guidelines are established.

a) Standardization of illumination source and test procedures for absolute radiometric calibration on the ground will be established to provide high accuracy and repeatability of data for an instrument. A goal is to establish techniques to provide correlation of data between instrument types.

b) On-orbit calibration will consist of a relative approach (self-contained internal sources) and an absolute check (sun calibration). No electronic calibration is recommended since this tends to add a potential noise source and is used mainly for trouble-shooting, not as an operational necessity.

c) DC restoration of each detector's output within the instrument is a design recommendation. This preserves dynamic range and reduces auxiliary data requirements and processing load on the ground - particularly at remote processing stations.

d) Two calibration targets shall be provided for the thermal band. If the instrument housing is used as a black body calibration source, its temperature shall be measured with appropriate sensors. These sensor calibration signals, and any other temperature calibration signals, shall be provided both to the house-keeping telemetry and inserted into the video signals for each scan line.

2.1.4.5 Sun View Protection

Above the normal range of radiation into the radiometer, under certain conditions, the radiometer may scan through the sun on several successive scans. The instrument shall be capable of scanning direct solar input without damage or reduction in life time. The radiometer shall return to its calibrated condition within one scan after exposure to the sun.

Sun shields may be used on the radiometer if the contractor requires them to supplement the internal optical baffles in suppressing unwanted radiation. If sun shields are used they shall be limited to the same envelope constraints as the radiometer. The sunshields are to be mounted as part of the radiometer.

2.1.5 Geometry

2.1.5.1 Mapping error budget

Geometric accuracy is the most critical system output product performance parameter. The baseline TM geometric mapping error budget required to meet these system accuracies, is shown in Table 4. This represents a large apportionment of the total system requirement.

2.1.5.2 Number of Detectors

The number of detectors per band shall have an interger relationship between the thermal channel and the other five bands. This aids in band-to-band registration and simplifies the hardware provided elsewhere in the system for compacted data modes.

2.1.5.3 Detector Registration

Along-Track (Cross-Scan)

Along-track registration errors between corresponding detectors of each band and from detector to detector within a band shall be $\leq .1$ IFOV.

Table 4. Geometric Mapping Error Budget Baseline

<u>Line Scanner</u>

Start of Scan Stability	Angular	-	3 μrad
	Time coherence to S/C clock	-	TBD
* Along Scan Positional Accuracy (repeatability along entire scan including optical distortions)			4 μrad
Across Scan-Non linearity (perpendicular to scan line)			4 μrad
Detector Position			
Placement (to a specific location)			.1 IFOV
Knowledge			0.05 IFOV

* Variations from this accuracy which are linear are acceptable.

Cross-Track (Along scan)

Cross-track registration errors from detector to detector within a band shall be $\leq$ .1 IFOV. Cross-track registration errors between corresponding detectors of each band should be minimized and shall be equal to an integer number of IFOV's. If cross-track band-to-band misregistration exists, the thermal channel shall be read-out first to reduce the storage required for registration during processing.

2.2 Electrical

2.2.1 General

The anticipated electrical input signal and output signal parameters, the power requirements and the electrical performance characteristics of the radiometer are given in this section. All specified characteristics shall be within tolerance over the lifetime of the equipment despite the combined effects of signal, impedance, and power supply variations (within specified tolerances, but taken at worst case values), radiation degradation, and environmental extremes.

2.2.2 Input Signals

The presence of any, all, or none, of the input signals applied in any sequence shall not damage the equipment, reduce its life expectancy or cause any malfunction, either when the radiometer is powered or when it is not.

2.2.2.1 1.6MHz Timing Signal Input

A buffered 1.6MHz balanced output is provided to the instrument and shall be used as the standard clock input to drive scan mechanisms and provide a reference frequency for internal timing. The clock signal is derived from a 3.2 MHz temperature compensated oscillator. The reference stability is $\pm$ 1 x 10^8 per year.

2.2.2.2 Command and Telemetry Interface

A remote decoder/mux will be located within the thematic mapper to provide the

command and telemetry interface between the instrument and the spacecraft. The command and telemetry data is distributed and collected using party lines and a 2 bus system to minimize the electrical interfaces between subsystems while maintaining high reliability.

Each remote contains sentry logic which checks each word on the request bus for its address. If its address is recognized, it turns on power to the remainder of the logic in the remote needed to perform the function. The supervisory word can request one of five functions: (1) pulse command execution; (2) serial magnitude data transfer; (3) analog data recovery; (4) bilevel digital recovery; (5) serial digital data recovery. A pulse command energizes one of 64 logic level outputs which remains energized for 20 msec. The remote then generates a status word for return to the telemetry format generator (TFG) that indicates the status of the remote response. A serial magnitude data transfer word activates one of four outputs which enables a user to accept data and clock for sixteen bits of information contained in the data word. Again, the remote sends a word back to the TFG on the response data bus to indicate parity check and status. An analog word activates one of 64 user inputs to the remote. This input will contain an analog signal which is digitized to eight bits and then transferred to the TFG on the response data bus along with parity and status information. A bilevel digital word activates eight (of the 64) sequential user inputs which contain bilevel digital signals. These inputs are formatted into an eight bit word and returned to the TFG along with parity and status information. A serial digital data word activates one of sixteen outputs and a clock which enables the instrument to supply eight bits of serial data to one of the 64 available remote inputs for transfer back to the TFG.

Data rates $\leq$ 8Kbps can be accommodated by the data busses and the STDN transponder.

Table 5 lists the performance of the command decoder and Table 6 indicates the telemetry mux interfaces with the instrument. All inputs circuit in the remote have an input impedance of 10 megohms minimum during non-sampling periods and 10 Kohms minimum during sampling. The input is capable of surviving s short circuit to + 28 VDC. Table 7 provides the size, weight and power of the remote that the instrument has to accommodate.

Table 5: Remote Command Decoder Outputs - 64 Total

Pulse Commands

Pulse Duration	20 ms minimum
Logical "1"	+5V @4ma
Logical "0"	0 to + .5V
R Source @ "0"	8.0K ohms maximum

Magnitude Commands

*Clock Rate	1.024 Mbps
*Gate Width	Envelopes 16 clock pulses
*Command Word	16 bits serial

* These signal outputs have the same voltage and impedance characteristics as those shown for pulse commands.

Table 6: Remote Telemetry Mux Inputs - 64 Total

Analog Inputs (Digitized to 8 bits)

Range	0 to +5 VDC
Z Source	5K ohms maximum
Accuracy	$\pm$ 30 MV

Bi-Level Digital Inputs

Logical "1"	+5 VDC
Logical "0"	0 $\pm$ 0.5 VDC
Fault Tolerance	-20 to +40 VDC
Z Source	5K ohms minimum; 10K ohms maximum

Serial Digital Inputs (8 bits/word) - 16 Maximum

Input Data	
Logical "1"	+5 VDC
Logical "0"	0 $\pm$ 0.5 VDC
Z Source	5K ohms maximum
Output Data	
Clock rate	1.024 Mbps
Gate Width	Envelopes 8 clock pulses

Outputs shall have the same voltage and impedance characteristics as those shown for serial magnitude commands.

Table 7: Remote Decoder/Mux Characteristics

Size	6 x 4 x 2 inches
Weight	1.5 lbs.
Power	1.2 watts from instrument

2.2.3 Power Subsystem

2.2.3.1 Description

The EOS power subsystem is a regulated direct energy transfer system which provides a +28 ± 0.3 VDC output. Regulation is obtained by controlling discharge boost converters, battery charge regulators, and a partial shunt regualtor. Separate power lines are provided to each user subsystem. Current sensing and protection are provided within the power module for each power output line.

2.2.3.2. Power Subsystem Characteristics

The power subsystem characteristics are delineated in Table 8.

Table 8: Power Subsystem Characteristics

Parameter	Value
Voltage	
Operating	+28 ± 0.3 VDC
Fault	≤ +45 VDC @ 100 μv/sec ≤ -10 VDC @ 250 μv/sec
Output Impedance	≤ 0.1 ohms, DC to 10 kHz
Noise (Output)	≤ 100 mv p-p
Load Transients	≤ +2 VDC @ 100 μv/sec
Line Drop	≤ 280 mv @ 100 W ≤ 500 mv @>100 W
Bus Protection	Contained in Power Module

2.2.3.3 Required Interface Information

Compatibility with the power subsystem requires that the instrument information listed in Table 9 be provided.

Table 9. Instrument Power Information Required

Parameter	Value/Limit
Power Demand (By Mode)	TBD
Power Input Filter Schematic	"
Grounding Diagram	"
Noise (Feedback)	"
Instrument Stabilization Time	"
Current Transients (Amplitude and Rise Time)	"

2.2.3.4 Instrument Design Criteria

The Thematic Mapper shall be designed to use $\leq$ 100 watts orbit average power including:

a) Warmup time

b) 15% On Time (Scanner and Electronics)

c) Temperature Compensation

d) Remote Decoder/Mux (1.2 watts)

e) Wideband Data Mux (TBD Watts)

2.2.3.5 Instrument Power Converter

A dc-to-dc power converter in the TM subsystem shall provide all voltages which may be required. In case of failure of the synchronizing circuits, the converter shall continue to operate in a free running mode. If more than one

dc-to-dc converter is required, they shall operate at frequencies which shall prevent the introduction of undesirable beat frequencies in the pass band of the radiometer.

2.2. 4 VIDEO OUTPUT DATA

The input to the spacecraft multiplexer (which adds auxiliary data and generates the composite digital data stream) shall be serial digital data on a per band basis. The Remote Encoder/Multiplexer is to be located within the instrument module and powered from the instrument power supply.

The data from each detector of the TM shall be supplied to the digital signal processor in analog fashion. In addition to these analog signals, the line synchronization signals shall also be routed to the signal processor on a separate buffer isolated line. A signal timing document shall be supplied by the Contractor delineating the specific times the various signals will be presented to the signal processor. Short circuit protection of all outputs shall be provided. The radiometer shall operate within specification, except for any shorted output while that output is shorted, and the entire radiometer shall operate within specification after removal of a temporary short.

The TM shall provide the following digital signals to the remote encoder/ multiplexer. A short to power bus or ground shall not cause any permanent damage to the instrument.

1. Start of scan or phasing pulse
2. End of scan pulse
3. Additional angular position data required to linearize scan
4. Instrument mode indicators that require sampling changes

Additional signals not required to encode the video data, but required to reconstruct and correct the TM images on the ground shall be provided through the telemetry remote. All ancillary data (timecode, corrections, modes) necessary for ground processing will be multiplexed into the composite video within the wideband data subsystem.

Table 10 indicates the remote encoder/mux characteristics.

Table 10. Remote Encoder/Mux Characteristics

Parameter	Value
Analog Inputs	DC Restored Each Scan Line
Number	100
Range	0 to 4.0 Volts
Input Impedance	TBD
Frequency Response	
Timing	
Signal Rise & Fall Times	
Digital Inputs (Mode & Synchronization)	
Number	
Range	
Input Impedance	
Digital Outputs	
Number	
Level	
Impedance	
Quantization Level	
Sample Rate	
Form Factors of Remote	
Size	
Weight	
Power	TBD

2.3 MECHANICAL

2.3.1 WEIGHT

The maximum weight of the entire TM subsystem (optics, electronics, cooler, insulation, etc.) shall be less than or equal to 160 kg (352 lbs).

2.3.2 ENVELOPES

Allowable Instrument envelopes are determined by the available launch vehicle dimensions for the fixed mounting configurations, and by the module storage and handling limitations for the modular arrangements. The maximum envelope shown in Figure 2 should be used as a guide in instrument design. Note that this envelope includes all instrument peculiar appendages including coolers and retracted aperture covers.

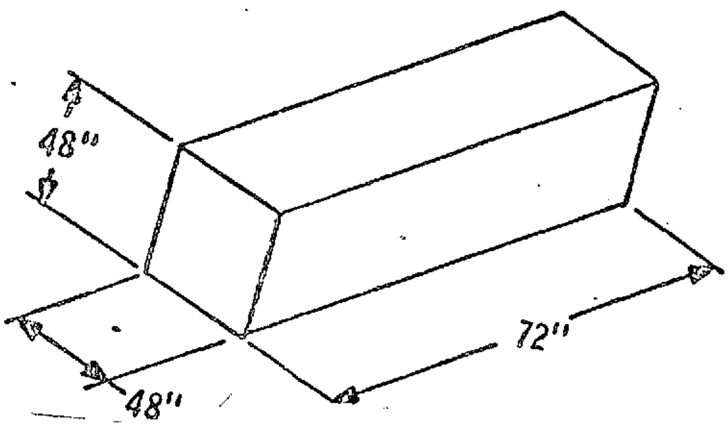

Figure 2. Instrument Maximum Envelope

Instrument aperture and cooler fields of view will be defined by the instrument contractor and will be determining factors in final instrument placement and module or support structure design.

Weight, center of gravity, and mass properties of each instrument will be supplied by the instrument contractor.

2.3.3 MOUNTING

A 3-point trunion mounting system (Figure 3) through the CG of the thematic mapper is suggested since it provides determinate planer reactions and minimizes induced strain into the instrument. Alternative mounting schemes can be accommodated, but must be negotiated with GSFC and the prime contractor.

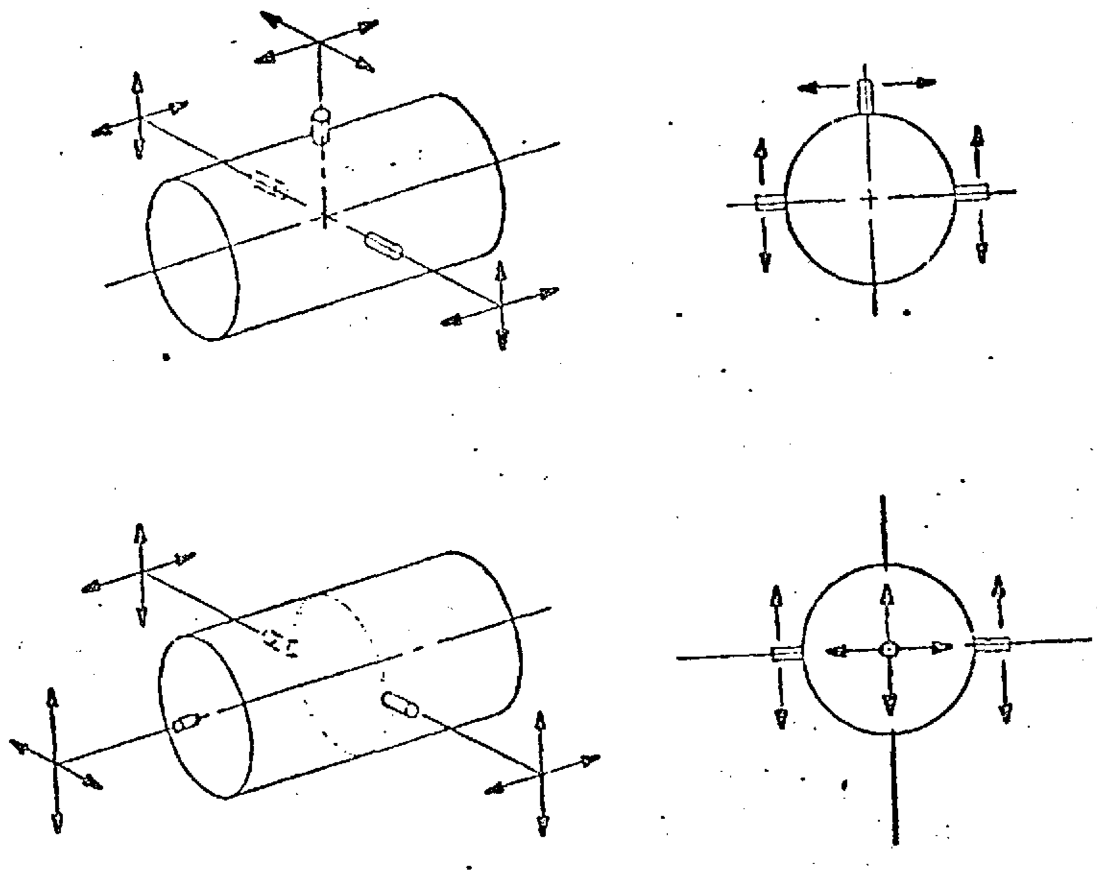

Figure 3. Three Point Reaction Systems

2.3.4 THERMAL CONTROL

Each module or instrument compartment will be designed to provide thermal isolation for the instrument and either the instrument or module will be individually insulated and have independent external heat rejection provisions. The instruments shall, in general, radiate heat from the earth viewing surface and coolers will be oriented outboard on the spacecraft anti-sun side as shown in Figure 4.

Instrument temperatures will be maintained by multi-layer insulation and passive heat rejection if possible; however, guard heaters may be required for a particular installation.

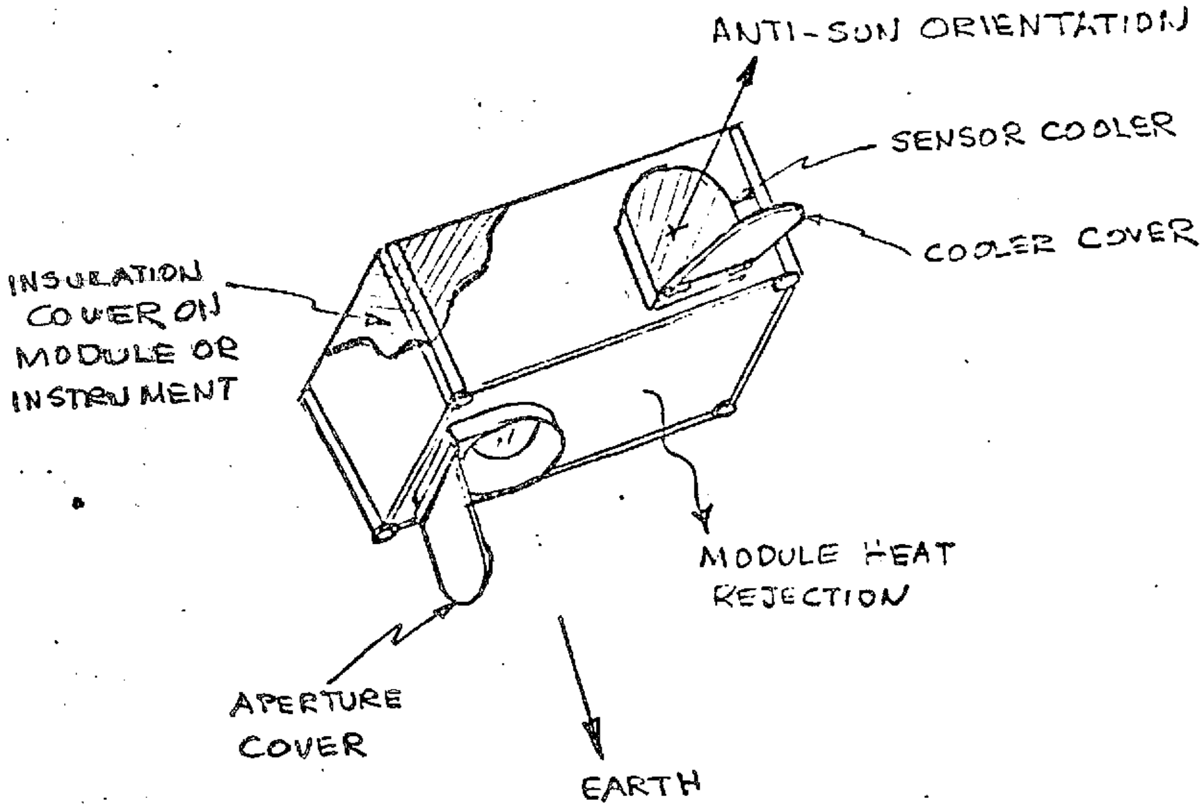

Figure 4. Instrument Thermal Control

The radiometer shall be designed to operate over a temperature range of $20 \pm 10^{\circ}C$.

A temperature analysis of the TM shall be conducted to determine heat transfer, including transient and steady state temperatures, and thermal gradients. This analysis shall include the degrading effects on image quality.

The thermal design shall include a passive detector cooler. The cooler will be mounted on the anti-sun side of the vehicle and shall look into a hemisphere except for the 60° cone subtended by the earth, and must be shaded from earth radiation. Laboratory means for cooling the required detectors will be acceptable during ground testing.

2.3.5 SCANNER ALIGNMENT

A method shall be provided for measurement of the alignment of the radiometer's optical axis with respect to two external reference surfaces on the radiometer frame. Provisions shall be made to precisely attach alignment mirrors and witness mirrors for contamination measurements either permanently or when required.

The alignment mirrors in turn will be used to boresight the radiometer to a mounting fixture and must be located in a convenient position for this purpose. The measurement of the optical axis alignment with the mirrors shall have an accuracy of ± 0.1 milliradians. Alignment of the axes of the six bands shall not change more than 0.1 milliradian as a result of all testing.